X-625-70-203

NASA TM X 65423

COMPREHENSIVE STUDY OF VENUS BY MEANS OF A LOW-COST ENTRY-PROBE AND ORBITER MISSION-SERIES

J. E. AINSWORTH

JUNE 1970



GSFC

GODDARD SPACE FLIGHT CENTER
GREENBELT, MARYLAND

N71-16807
(ACCESSION NUMBER)
110
(PAGES)
TMX-65423
(NASA CR OR TMX OR AD NUMBER)
(THRU)
63
(CODE)
31
(CATEGORY)
FACILITY FORM 602

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J. E. Ainsworth

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ACKNOWLEDGMENTS

Many people have contributed to the material presented here.

The Planetary Explorer bus vehicle and the Fields, Particles, and Aeronomy Orbiter mission were developed at GSFC under the direction of P. G. Marcotte.

The use of the Delta rocket and Planetary Explorer bus for a multiprobe meteorological mission to Venus was first suggested and studied by C. C. Stephanides at GSFC and additional study under his direction was carried out under contract with the AVCO Corporation.

The Mapping Orbiter has resulted in large part from the orbiter radar study done by H. J. Goldman and D. L. Roberts at IITRI.

The Balloon, Seismic Lander, and Television Lander missions were developed as a company sponsored effort directed by S. J. Ducsai of the Martin Marietta Corporation.

I wish to express my personal appreciation to the many scientists and engineers at GSFC and throughout the country whose interest and assistance have sustained the effort of the past three years.

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PART I

DOCUMENT OBJECTIVES

1. BACKGROUND

Recent technical accomplishments and studies presage the beginning of a period of greatly increased interest in the exploration of Venus. Three years ago Venus orbiters were known to be feasible but there was concern that descent-probes would not outlive atmospheric entry. If descent-probes could indeed survive entry, it seemed apparent that the high atmospheric pressure and temperature near the surface and the rapid descent required to accomplish relay communication via a fly-by bus would limit probe use to pressure, temperature, and simple optical measurements. A year later it was clear that not only was it possible to survive atmospheric entry but that a number of sophisticated atmospheric and cloud physics experiments could be performed during slow descent to the Venus surface. Still later, less than a year ago, it became apparent that relatively simple vehicles could be designed which would allow useful measurement on the Venus surface by seismic and other instruments. At about this same time Earth based microwave interferometer measurements of the polar temperature, and the U.S.S.R. Venus 5 and 6 measurements determined that both polar and mountain top temperatures were too hot to allow Earth forms of life to survive on the Venus surface. The result of these findings is a reduction in the effort needed to meet international sterilization requirements.

In less than three years we have advanced from a condition where poor prospects existed for making the in situ measurements necessary for a significant advance in our understanding of Venus to a condition where it is possible to apply a substantial number of the most efficient techniques and instruments used in four major areas of planet research: aeronomy, meteorology, planet surface examination, and planet-interior study. In some instances the fabrication and use of the instruments for Venus will present less difficulty than is involved in their use on missions to Mars and other planets. In other instances, the special requirements of Venus use will stimulate the additional instrument development contemplated for improved Earth use.

2. MISSION DEVELOPMENT

Present and past Venus mission studies have involved:

a. Delta launched payloads of 800 pounds.

b. Atlas-Centaur launched payloads of 1850 pounds.

c. Titan launched payloads of 3900 pounds.

This document will consider the spectrum of uses of the 800 lb. Delta payload.

The first phase of the study of Delta payload uses is an attempt to select one of each of the different classes of missions necessary for a comprehensive study of Venus and to show both the feasibility of each class and the scope of its resulting Venus examination. The classes of missions included in this first phase of the study are:

a. Aeronomy and meteorology using an entry bus and descent probes.

b. Orbiter use for field, particle, and aeronomy measurements.

c. Atmospheric circulation measurement by means of balloon probes.

d. Orbiter use for physical, thermal, and optical mapping of Venus.

e. Venus interior examination by means of both active and passive seismic experiments.

f. Venus surface examination by means of a landed probe.

These missions and their state of development are described in Part II.

The second study phase must be to define a series of missions in which the design of each individual mission is such as to insure the maximum science return from the series rather than from the individual mission.

3. OBJECTIVES

The purpose of this document is three-fold:

a. To acquaint the scientific community with the concept of a comprehensive study of Venus by means of a low-cost entry probe and orbiter mission-series,

b. To solicit scientific community assistance in further development, modification, and refinement of the concept, both in part and in its entirety,

c. To solicit scientific community assistance in clearly and completely stating the rationale for the concept. i.e. The merit and priority of the concept in the context of general planet studies; as related to an understanding of Earth environmental and pollution problems; as related to past and future flight and Earth based Venus measurements; as a stimulus to the further development of needed Earth measurement instruments; as related to international participation; and as related to alternate Venus exploration concepts.

PART II

MISSION-SERIES DESCRIPTION

1. LAUNCHING

1.1 LAUNCH VEHICLE

The launch vehicle chosen for the low-cost mission-series is the Delta, see Figure 1. This launch vehicle,* which was first flown in 1960, has been subject to a continuous extension of its capability by means of a series

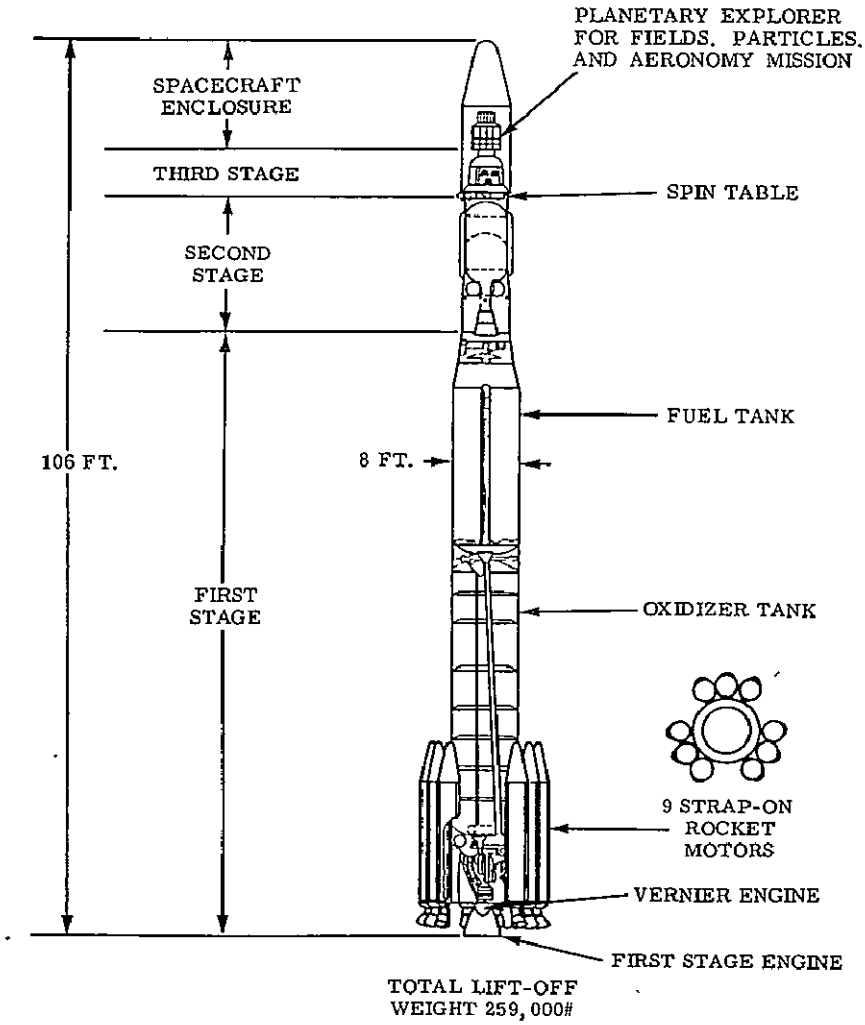


Figure 1. Delta Launch Vehicle

*Delta LTD/9 Castor II/AJ10-118F/TE-364-4.

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of thoroughly tested incremental steps and as a result has maintained a record of exceptional reliability. Approximately eleven missions will use the chosen launch vehicle between 1971 and 1975 when its first use on the proposed Venus mission-series would take place.

The liquid-fuel first stage engine is gimbal-mounted to provide attitude control from lift-off to first stage engine cut-off. Two liquid-propellant vernier-engines provide spin control throughout first-stage operation, and attitude control from first stage engine cut-off to first stage separation. Nine strap-on solid fuel rocket motors assist the first stage.

The liquid-fuel engine of the second stage is likewise gimbal mounted to provide attitude control throughout the second stage burn. A nitrogen gas system is used to provide second stage spin control during powered flight and both spin and attitude control after engine cut-off. The nose fairing, which protects the payload from aerodynamic heating and biological contamination, is jettisoned at 125 km shortly after the beginning of second stage burn. Two fixed helium-nozzles, fed by the propulsion pressurization system, provide second stage retro-thrust following separation of the third stage.

The third stage is supported on the second stage by means of a rotation table. Following cut-off of the second stage engine, the table is spun by solid fuel motors affixed to its perimeter. The spin stabilized third stage is then separated from the second stage and the solid propellant third stage motor is ignited.

An inertial guidance system, consisting of an inertial sensor package and a digital guidance-computer, controls both the vehicle's motion and its sequence of operations from lift-off to third-stage separation. To control vehicle motion, the sensor package provides vehicle attitude and acceleration information to the guidance computer which computes the vehicle's position, velocity, and attitude, compares the computed values with stored reference values, and then generates vehicle steering-commands to correct for deviations.

Venus mission-series launches will be from Cape Kennedy, Florida. The second stage will place the launch system in a nearly circular parking orbit at an altitude of about 170 km. Completion of a parking period of 10 - 30 minutes, as determined by the particular mission, will be followed by third stage spin up, third stage separation, second stage retrothrust, and the third stage burn which accomplishes escape-velocity and injection into an Earth-Venus transfer orbit. The Planetary Explorer bus is then separated from the third stage. At this point orbit error is such that there is a negligible possibility that the spent third stage will be captured by Venus.

1.2 LAUNCH VEHICLE CAPABILITY

Analysis has revealed that for a given launch date and orbit injection-energy, and for less than one revolution around the sun, there are as many as four different flight paths to Venus.* For the proposed mission-series, however, launchings cannot be scheduled for an arbitrary date but must be restricted to a "window" of several weeks when the relative positions of Earth and Venus allow the orbit injection velocity required by the payload to be furnished by a launch vehicle of practical size and cost, and allow satisfactory communication and entry conditions to be obtained at arrival. Favorable launch windows occur approximately 2.5 months before each closest approach, or inferior conjunction, of Earth and Venus and thus the windows are spaced by the "synodic" period of approximately 19.2 months. Although missions using the same launch vehicle and occurring during successive launch windows differ in their payload capacity and other Venus arrival parameters, mission conditions are cyclic. Every eight years, or every fifth Earth-Venus inferior conjunction, a Metonic cycle† is completed; approximately the same space fixed Earth-Venus geometry re-occurs; and the Venus arrival parameters duplicate those of the launch window eight years earlier.

Figure 2 portrays various spacecraft weights that can be sent to Venus by the proposed launch vehicle during a Metonic cycle. The data, which is for entry-probe missions, represents the maximum spacecraft weight associated with the indicated entry-speeds of the probes into the Venus atmosphere and is for ten day launch windows. The 36,000 ft/s entry speed‡ shown for the 1975 launch represents a conservative estimate of 1969 heat shield capability. For the remaining windows in the cycle it can be seen that higher entry speed and thus greater heat shield capability is required if we wish to maintain spacecraft weight at 800# or more. AVCO has stated that it is reasonably certain that the heat shield material selected for the 36,000 ft/s entry speeds in 1975 will be able to be used at entry speeds in excess of 42,000 ft/s. Confirmatory testing of this material, as well as the development of new materials, seems certain to furnish the 38,500 ft/s entry capability necessary for the 800-840# spacecraft weights of the proposed mission series.

*Type I paths are those having heliocentric path-angles of less than 180 degrees. Type II path angles are greater than 180 degrees. Because the heliocentric motions of the Earth and Venus are not coplanar, two paths usually exist for each path type, and are designated as class I and class II. Class I paths have shorter flight times and smaller heliocentric path-angles than Class II. Thus, the type I, class I path has the shortest flight time, and the type II, class II path has the longest flight time.

†Meton, a 5th Century B. C. Greek astronomer, discovered that a period of 19 years elapses before the full moon occurs on the same day of the year. This fact became the basis of the Greek calendar.

‡Entry speeds given here will occur at a distance of 6150 km from the center of Venus.

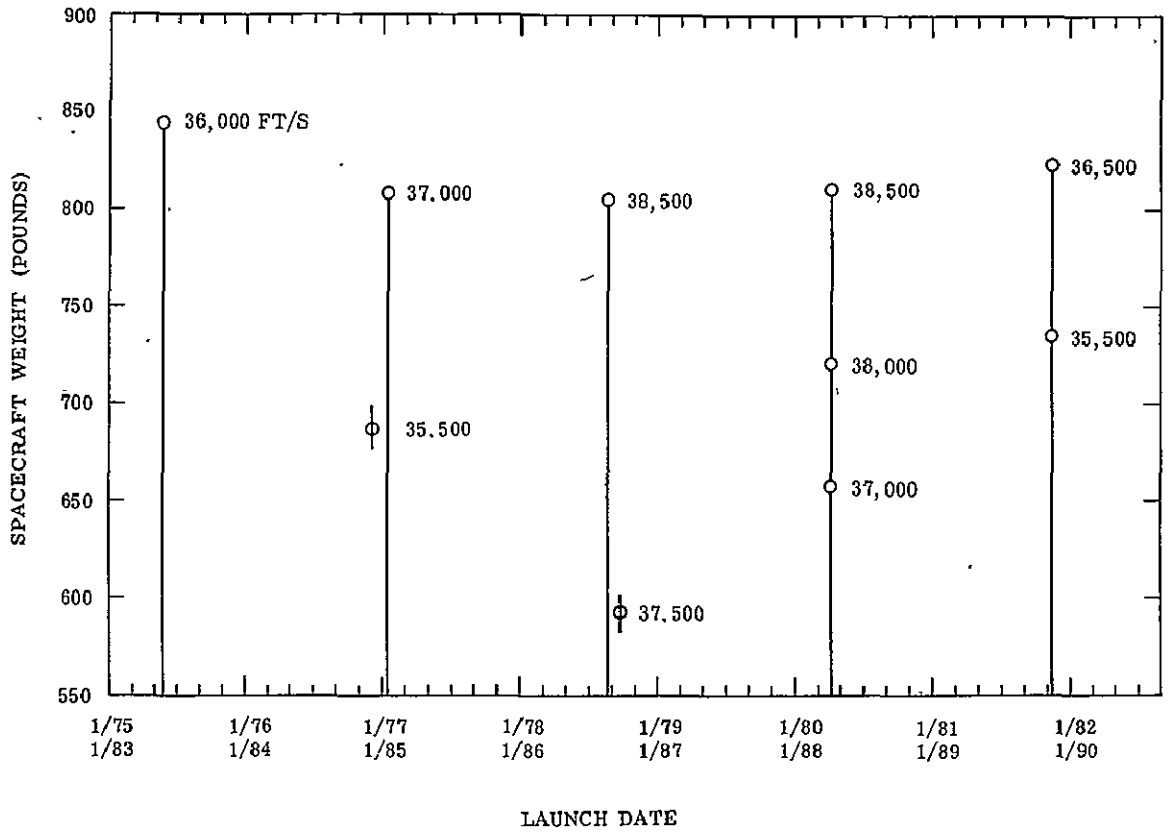


Figure 2. The maximum spacecraft weight that can be sent on entry probe missions to Venus by a Delta launch system is shown in relation to the speed of a probe entering the atmosphere 6,150 km from the center of Venus.. Probe speeds are in ft/s. Launch windows are 10 days.

Other launch and arrival-parameters such as launch geometry, Venus approach geometry, and Venus-Earth communication distance are involved in specifying the missions shown on Figure 2 but do not present significant constraints to the selected mission-series.

The weight which can be placed in orbit around Venus will be discussed in the Section 2.4.

The continuing effort to improve the Delta launch system can be expected to result in increased payload capability. The most recent change, an improved nozzle on the second stage, will result in an additional 25 pound capability which has not been included in the mission-series described here.

2. EARTH-VENUS TRANSIT AND VENUS ORBITING

2.1 PLANETARY EXPLORER BUS VEHICLE

The Planetary Explorer (P. E.) bus is based upon Phase A design studies by the Goddard Space Flight Center (Reference 1) and AVCO (Reference 2) and is shown in its various mission-series configurations in Figures 4, 7, 15, and 21.

The bus is expected to be suitable for use during long missions because it is spin stabilized and is thus conservative of fuel, both in transit and in orbit. It is expected to be reliable because of its simplicity. It is expected to be low-cost due to both its simplicity and to the fact that its basic design allows easy conversion to meet the bus requirements of a variety of probe and orbiter missions.

2.2 TRANSIT DESCRIPTION

During transit, maintaining good communication with Earth and optimum use of the bus solar cells require that the bus spin-axis remain perpendicular to the direction of the sun. Spin stabilization alone is sufficient to maintain this perpendicularity throughout transit if the spin-axis is placed perpendicular to the transit orbit-plane as shown in Figure 3.

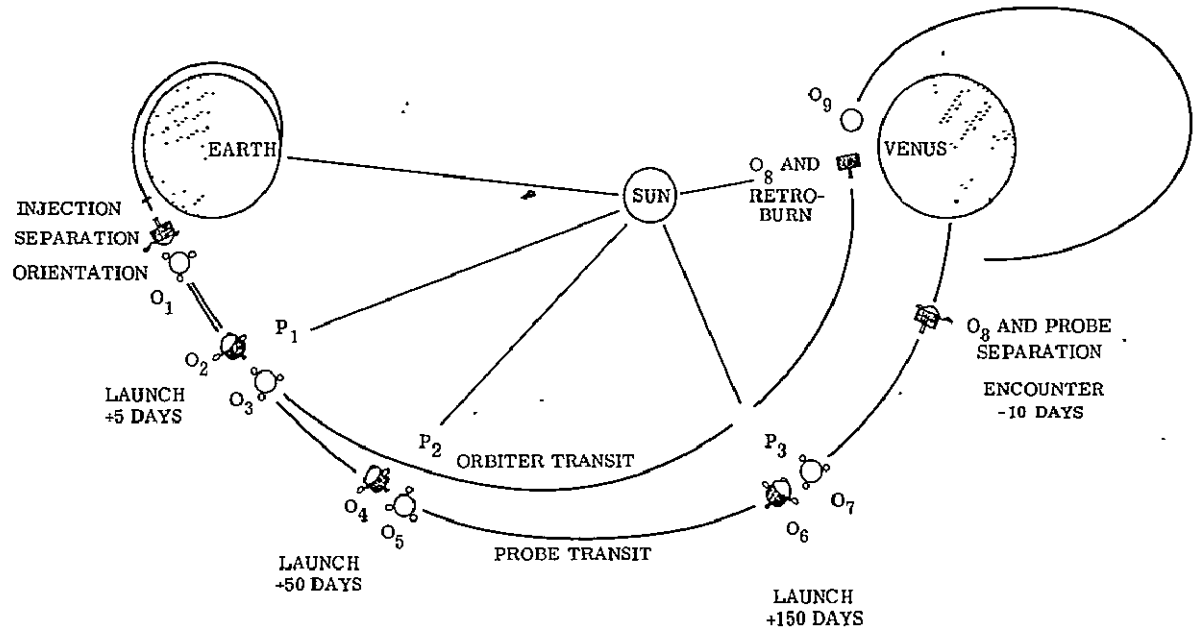


Figure 3. A schematic view of bus transit as seen from the north celestial pole.

Following bus separation from the third stage of the launch system, the bus is spinning at 30 rpm* and its spin axis lies in the transit orbit plane. The first bus maneuver, 0_1 of Figure 3, places the spin axis perpendicular to the transit orbit plane. Subsequent orientation maneuvers take place immediately before and after course-correction-pulse applications at P_1, P_2, and P_3. An example of aim point errors in the target plane following injection and course correction pulsing is shown in Table 1 for a Type II, 1975 transit.

Table 1

Aim-point Errors for a 1975 Type II 176 Day Transit

Event	Time from launch (days)	Velocity change (m/s)	Resulting 99% dispersion ellipse semi-axes (km)	
			a	b
Injection			4,570,000	76,600
First correction	5	100	85,900	8,200
Second correction	50	7	9,520	381
Third correction	150	4	831	316

Although the geometric capture-radius of Venus is roughly 6,100 km, gravitational focusing at Venus extends the aim-point capture-radius in the target plane to 16,000 km for the 1975 transit. Table 1 shows that for this mission, capture of the bus by Venus can be assured following the second course-correction. Gravitational focusing reduces the aim error given in Table 1, and for the case where the aim point is near the center of Venus a reduction of from two to five will be obtained.

*Third stage spin is at 70 rpm. Extending of the probes, see Figures 4 and 9 reduces bus spin to 30 rpm.

2.3 BUS SUBSYSTEMS DESCRIPTION

The basic bus, see Figure 4, comprises essential subsystems which are attached to a load bearing central support tube, and which can be conveniently modified or added-to according to mission requirements. The subsystems are:

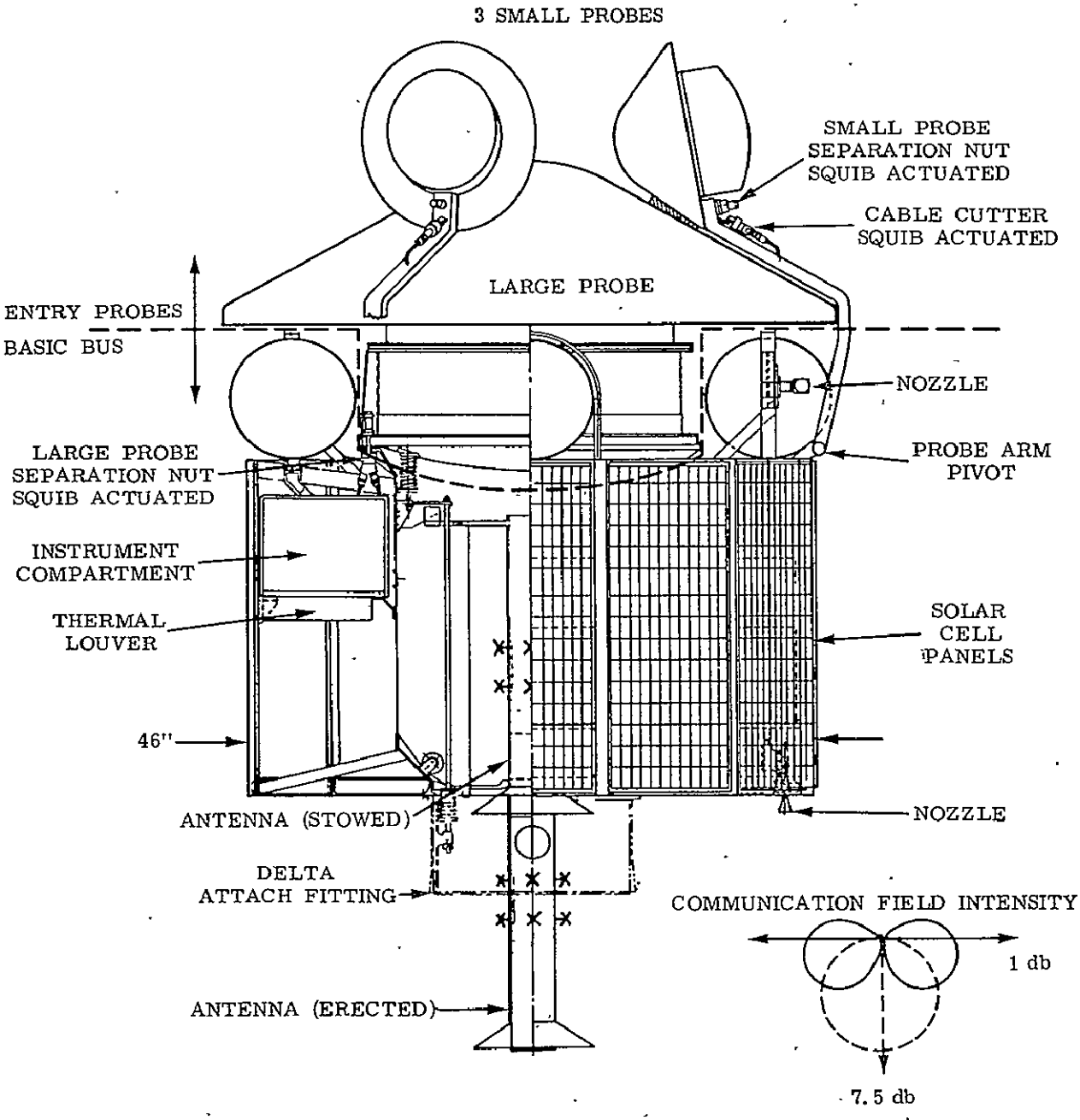


Figure 4. The basic bus structure adapted to carrying four meteorology entry probes. The left side of the bus is shown in cut-away. The Delta attach fitting remains with the third stage. The dual-pattern antenna is erected after third stage separation.

2.3.1 Power

The solar cell array provides the electric energy for the bus vehicle and feeds directly to a shunt regulated 28 volt distribution terminal from which all of the bus electrical systems are supplied. The battery is connected to the 28 volt distribution terminal through a charge and discharge regulator. While the bus is in sunlight the battery is continually charged but when the bus is shadowed or when temporary requirements for greater power exist, the battery replaces the solar cells in toto, or in part, to supply the bus electrical requirements. For the configuration shown in Figure 4, the solar cells furnish 65 watts at Earth and 122 watts in the vicinity of Venus. However, the solar panel area can be readily changed to satisfy greater or lesser needs. Table 2 lists the component weights of the power system.

2.3.2 Attitude Determination

Two principal units are used in attitude determination. The first unit is the scanning celestial attitude-determination system (SCADS), a star tracker which measures star magnitude and angle and feeds this data over the telemetry link to Earth. At Earth, computers are able to determine the direction of the bus spin axis with an error of 0.10-degrees. The second unit is the solar-aspect system, which gives the spin-axis sun angle and provides a timing pulse each time its sensor scans the sun. The timing pulses are used in conjunction with a sector generator and the command system to obtain proper synchronization of bus spin-axis-direction-change and course-correction propellant pulses during transit, and for proper probe release timing at Venus. The sector generator is also used for proper sequencing of measurements by bus science instruments when this requirement exists. There is, in addition, a planet sensor which can give a unique bus spin-axis direction when the spacecraft is near Earth or Venus.

2.3.3 Attitude Control and Course-correction

The basic system is used during transit, and can also be used during orbit around Venus, for bus spin-up, spin-down, spin-axis-direction-change, and course-correction. Typical total system capability is 130 rpm spin change, 720° of spin-axis direction change, and 125 m/s course-correction change. In order to separate rotation changes from translational motion seven thrust nozzles are used. The arrangement used satisfies all bus rotation requirements and allows bus course-corrections to be made in any direction within a

Table 2

Planetary Explorer Bus Weight Summary for a Probe Carrying Bus

Subsystem	Element Weight, lbs.	Assembly Weight, lbs.
Structure		52.3
Instrument shelf	6.1	
Center tube	10.0	
Booms and hinges	8.2	
Frames, fittings, etc.	36.0	
Power		40.3
Solar panels	22.1	
Battery	9.9	
Converters	4.0	
Regulation	2.8	
Probe battery charger	1.5	
Attitude Determination		13.5
Sun sensor and electronics	3.5	
SCADS and electronics	9.0	
Attitude control logic	1.0	
Propulsion and Orientation		99.3
Tanks (4) and pressurant	11.7	
Propellant (hydrazine)	70.0	
Nozzles, lines, etc.	13.0	
Damper	4.6	
Communications		26.1
Receivers (2)	6.0	
Transmitter	5.9	
Diplexer	0.7	
Antennas	4.0	
Decoder	3.0	
Memory (science)	4.0	
Memory, A/D converter, encoder, clock	2.5	
Programmer		4.0

Subsystem	Element Weight, lbs.	Assembly Weight, lbs.
Thermal Control		19.9
Blankets	7.6	
Coatings	2.0	
Louvers (8)	4.5	
Heaters and thermostats	2.0	
Heat pipes	3.8	
Cabling		13.0
Contingency		10.0
TOTAL BUS WEIGHT		278.4

hemisphere about the spin axis without the necessity for spin-axis direction change.* Maximum efficiency is obtained, however, by reorienting prior to course-correction and subsequently returning the spin axis to its original direction perpendicular to the ecliptic plane.

Pure-hydrazine propellant is stored in four symetrically spaced tanks shown in Figure 4. The 0.3 g developed at the propellant tanks due to bus spin forces the propellant against the outside of the tanks, establishing a centrifugally maintained barrier which permits the use of a bladderless† nitrogen-gas blow-down pressurization system. In the nozzle, the liquid hydrazine passes over a catalytic bed which causes a spontaneous exothermic decomposition into a nitrogen-hydrogen gas jet developing 5 pounds of force. The propulsion jets are commanded from Earth; they can operate continuously or in a pulsed mode with a pulse centroid repeatability of 1 milli-second. Normal operation would use 0.35 pound-second pulses with a 0.1 second duration. Spin axis precession introduced

*An 8th nozzle is necessary to obtain spherical translation capability without a 180° spin-axis direction change.

†In non-spinning "weightless" spacecraft the pressurant gas will pass through the propellant and will not force the propellant to feed into propulsion engine. The pressurant must be prevented from mixing with the propellant by means of a flexible container or bladder.

by spin-correction or course-correction, or resulting from probe separation, is removed by means of a passive damper having a time constant of 30 minutes. Nominal angle-error in bus course-correction and in probe release, from all causes, is approximately 1°.

2.3.4 Communications

Earth-bus communication involves links for three different activities:

a. The data link transmits 10 bit word housekeeping, engineering, bus attitude, and experiment data at 15 bits per second during cruise and at 315 bps* when the bus enters the outer atmosphere of Venus.

b. The command link is capable of 128 commands which can:

 i. switch the bus transmitter to 1, 5, and 10 watt outputs.

 ii. switch the transmitter output to either the 1 db or to the 7.5 db antenna;

 iii. order attitude and course-correction changes and probe releases.

 iv. order other required operations.

 The command data rate is 1 bps.

c. The tracking link determines the position and velocity of the bus at all times by means of range and Doppler measurements obtained from a turn-around transponder on the bus. Speed and range errors relative to Earth are 1 mm/s and 6 meters respectively.

All links use the Deep Space Network (DSN) 210 ft. diameter antennas. Command and tracking utilize 400 kw transmission from these antennas.

The erectable bus antenna, see Figure 4, obtains the hemisphere coverage needed for maneuvering by means of pattern switching.

*For the 1975 mission communication distances.

The 1 db low gain circularly polarized pattern is doughnut shaped with its principal axis coinciding with the bus spin axis and it thus looks at right angles to the spin axis and in all spin-azimuth directions. The 7.5 db high gain circularly polarized pattern is roughly spherical in shape and looks outward along the spin axis of the bus.

The 2300 MHz solid state S-band transmitter-receiver which performs the function of a turn-around transponder transmits a pulse code modulated/phase modulated (PCM/PM) convolutional encoded signal. The transponder uses redundant receivers in order to guard against command failure. Upon arrival at Venus communication distances for the mission series are from $60(10)^6$ km to $135(10)^6$ km, depending upon the particular mission opportunity.

2.3.5 Thermal Control

Because solar irradiation at Venus is twice that occuring in the vicinity of Earth, solar-panel temperatures of as much as 75°C may be experienced. The bus electronic instruments have an operating-temperature range of -10°C to 50°C and must be shielded from the solar cells by means of both insulation against conduction and by a multi-layer aluminized thermal radiation blanket. Changes in the internal energy dissipation by the electronic instruments and lack of a suitable heat sink require that a system of active variable-emittance louvers, see Figure 4, be used to keep the electronic instruments within their operating range as Venus is approached.

The bus nutation damper requires a temperature of greater than 22°C at all times and is placed at a location within the instrument compartment where this condition is obtained. The hydrazine tanks, fuel lines, and nozzles are coated with a paint having a ratio of absorbtivity to emissivity which maintains them at a temperature greater than their minimum operating temperature of 2°C at all times. Entry probes are insulated from the bus so that their release will not result in significant bus temperature changes.

2.4. VENUS ORBIT DESCRIPTION

For an observer on the Sun, the bus appears to be traveling in a direction inclined by only a few degrees to the Venus orbit plane. For an observer on Venus, however, the inclination of the bus approach assymptote is determined by bus velocity relative to Venus, rather than by the absolute bus velocity, and the inclination is considerably larger. For the proposed mission-series the inclination of the approach assymptote is roughly 30° for each of

the missions; the approach to the planet is from above the orbit plane for some opportunities and from below the orbit plane for others. In any case, it is this approach inclination which determines the minimum inclination of the bus orbit around Venus, see Figure 5*. The maximum inclination, 90°, is obtained when the bus is aimed so as to pass over the north or south celestial pole of Venus.

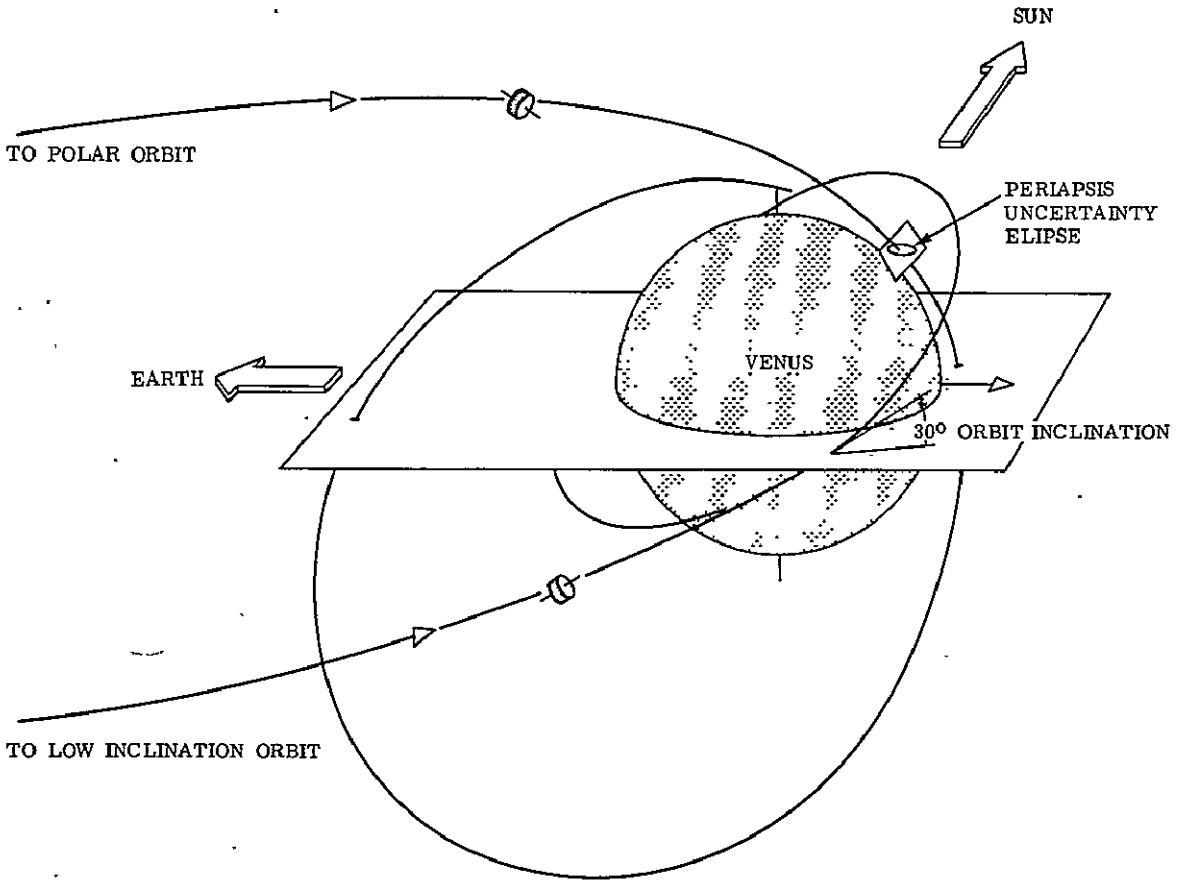


Figure 5. Geometry of typical orbits around Venus.

The periapsis altitude or closest approach point of the orbit is determined primarily by the bus aim point, and the uncertainty in periapsis altitude, see Figure 5, is determined by both the uncertainty of the position of the bus at the last course-correction and the course-correction error.† In order to

*Unless prohibitive additional bus propulsion is devoted to adjusting the approach assymptote direction.

†Gravitational focusing pulls the aim point toward the planet and reduces the aim point and course-correction errors.

obtain the greatest certainty in placing the bus into its desired orbit and to minimize the retro-motor propellant weight, retrofire takes place at the periapsis of the hyperbolic path of the bus and tangent to the Venus surface. The region of retrofire then becomes the periapsis of the orbit ellipse.

Satisfactory temperature control of the bus over long periods requires that its spin axis remain perpendicular, within 10°, to the bus-sun line. Greater angles are possible for short periods but care must be taken to prevent direct or reflected sunlight on the louvers. It is necessary to select an Earth-Venus transfer orbit and a Venus orbit for which this constraint is met during the retro-maneuver when the bus spin axis is tangent to the Venus surface.

For a given Venus-orbit periapsis-altitude the apoapsis altitude is determined by the bus asymptotic approach speed, which should be as low as possible in order to save retro-motor propellant weight, and by the magnitude of the retro-motor impulse. Since the science payload of the orbiting bus is the load capacity remaining after the addition of the retro-motor, apoapsis altitude can be related to the science payload in orbit as shown in Figure 6 for the fields, particles, and aeronomy orbiter.

2.5 ORBITING BUS SUBSYSTEMS DESCRIPTION

A typical fields, particles and aeronomy orbiter is shown in Figure 7. A weight breakdown is shown in Table 3.

2.5.1 Retro-motor

The 38 pound retro-motor case is fitted within the central support tube of the bus, see Figure 7. As much as 365 pounds of solid propellant can be used; the amount is governed by the desired apoapsis altitude and the load capability at launch. A 4800 pound retro-thrust is developed for a nominal burning time of 22 seconds.

2.5.2 In-Orbit Maneuver

The hydrazine spin-change and course-correction system is supplied with additional fuel for in-orbit spin-change, spin axis direction-change, and periapsis change (1500 km).

2.5.3 In-Orbit Communication

A data storage-unit is required for orbiting missions where the data accumulation rate is higher than the data transmission rate, where Venus may at times block line-of-sight transmission from the orbiter to Earth, or where Venus may block the Sun from the solar

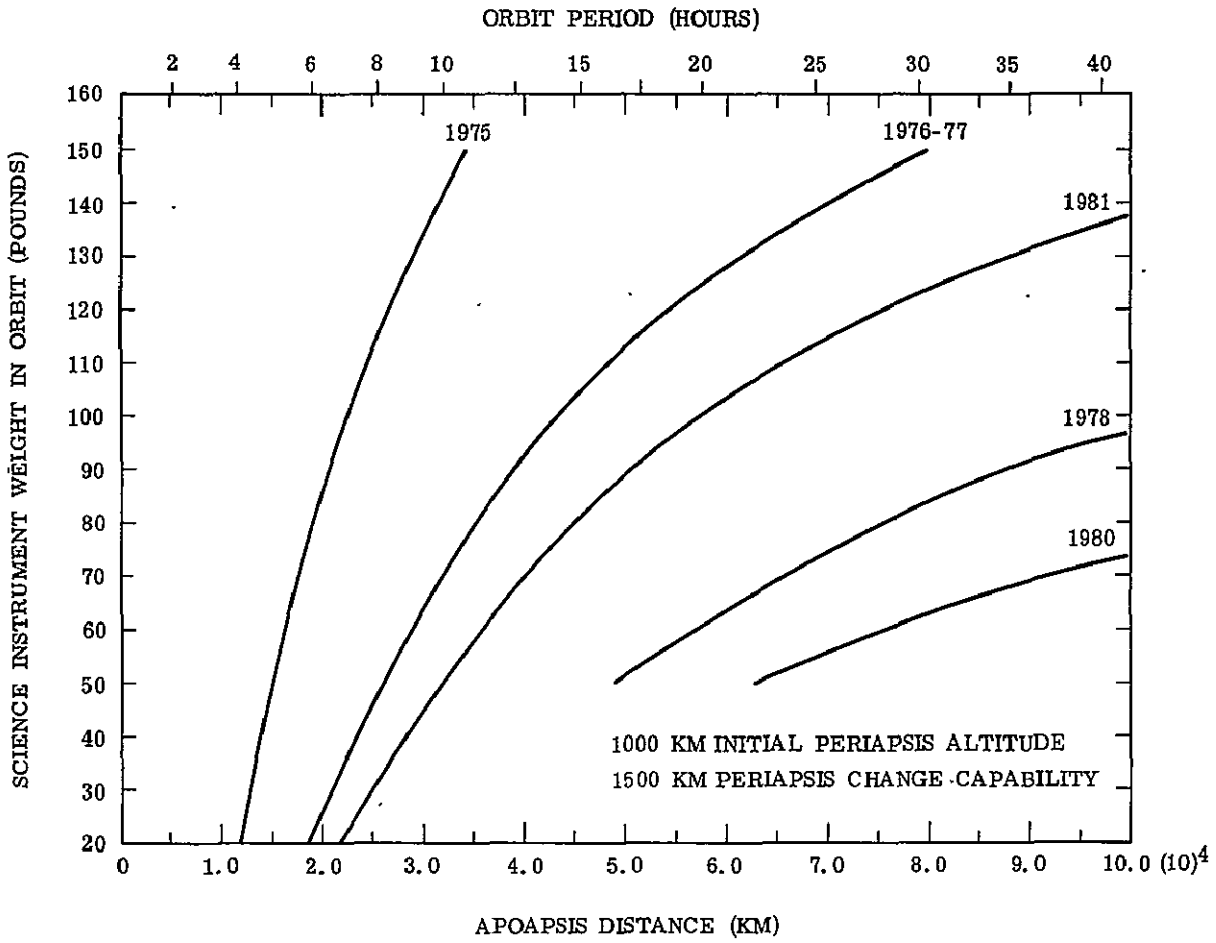


Figure 6. Averaged values for science payload in orbit around Venus as related to apoapsis altitude for a fields, particles, and aeronomy orbiter. Periapsis change can be obtained by in-orbit use of the attitude and course-correction system.

cell array and the supply of energy from the batteries alone is insufficient for simultaneous measurement and transmission. A twenty pound $5(10)^5$ bit storage-unit is available for use on these missions.

A second orbiter adjunct is a dual mode antenna, see Figure 7, consisting of 32 rows of 4 colinear dipoles mounted parallel to the bus spin axis. In the low gain mode the linearly polarized antenna radiation pattern is doughnut shaped with its principal axis coinciding with the orbiter spin axis. Antenna gain is 8 db, beam width is 15°; transmission rates to Earth are shown in Figure 8.

In the 19 db high gain mode, the 360° azimuth coverage of the doughnut pattern is shrunk to 10°. The resulting 10° by 15° unidirectional radiation pattern is electronically despun using the sun signal as an azimuth reference and is pointed to a fixed direction in the

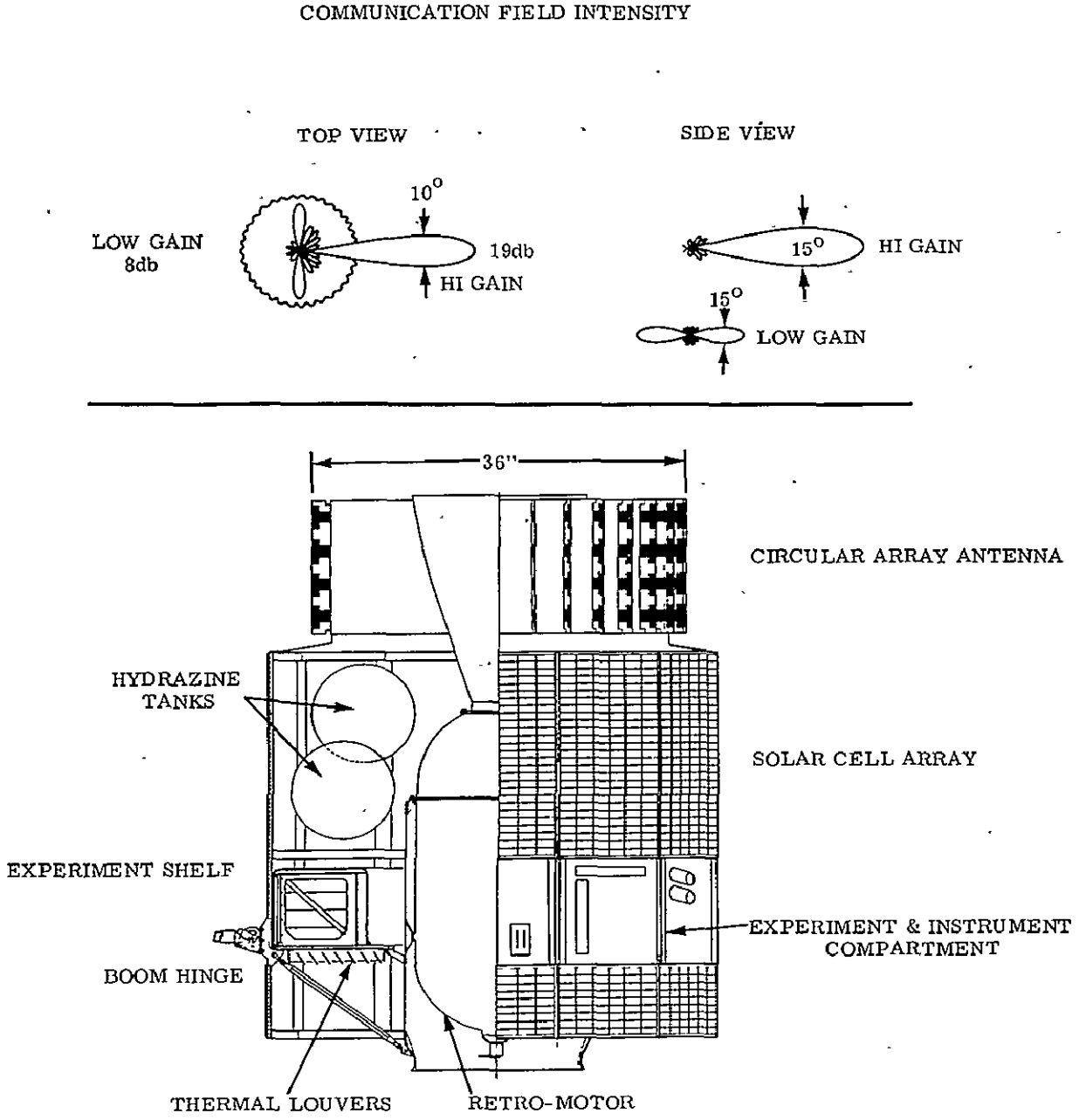


Figure 7. A Venus orbiter configuration suitable for fields, particles, and aeronomy experiments. The left side is shown in cut-away.

ecliptic plane. By Earth command, using the bus omni-directional antenna, the despun high-gain beam is changed in azimuth until it is directed at Earth. Subsequent shifts in direction of the beam are made as required by the change in Sun-Venus-Earth angle with time.

Table 3

Fields, Particles, and Aeronomy Orbiter
Weight Summary for a 1976-77 Mission

Item	Pounds
Structure	61.2
Experiment booms	12.0
Thermal system	19.5
Communications system	29.3
Data storage and encoding	24.5
Attitude-determination system	9.5
Power system	
Electronics and batteries	19.8
Solar array	22.1
Programmer instrumentation	4.0
Cabling	15.0
Planetary-insertion motor dry weight	38.2
Hydrazine system	
Fixed dry weight	14.6
Tanks and pressurant	13.3
Unusable propellant	2.0
Subtotal	285.0
Contingency (approximately 10% of subtotal)	23.0
Subtotal	308.0
Experiments (20% contingency)	50.0
Total in-orbit spacecraft dry-weight	358.0
Propellants	
Fixed attitude and spin control	11.0
Mission-dependent propellants	
Trajectory correction (in transit)	44.0
Orbit maneuver (in Venus orbit)	35.0
Planetary insertion solid propellant	285.0
Total spacecraft liftoff weight	733.0
1976-77 mission launch capability	733.0

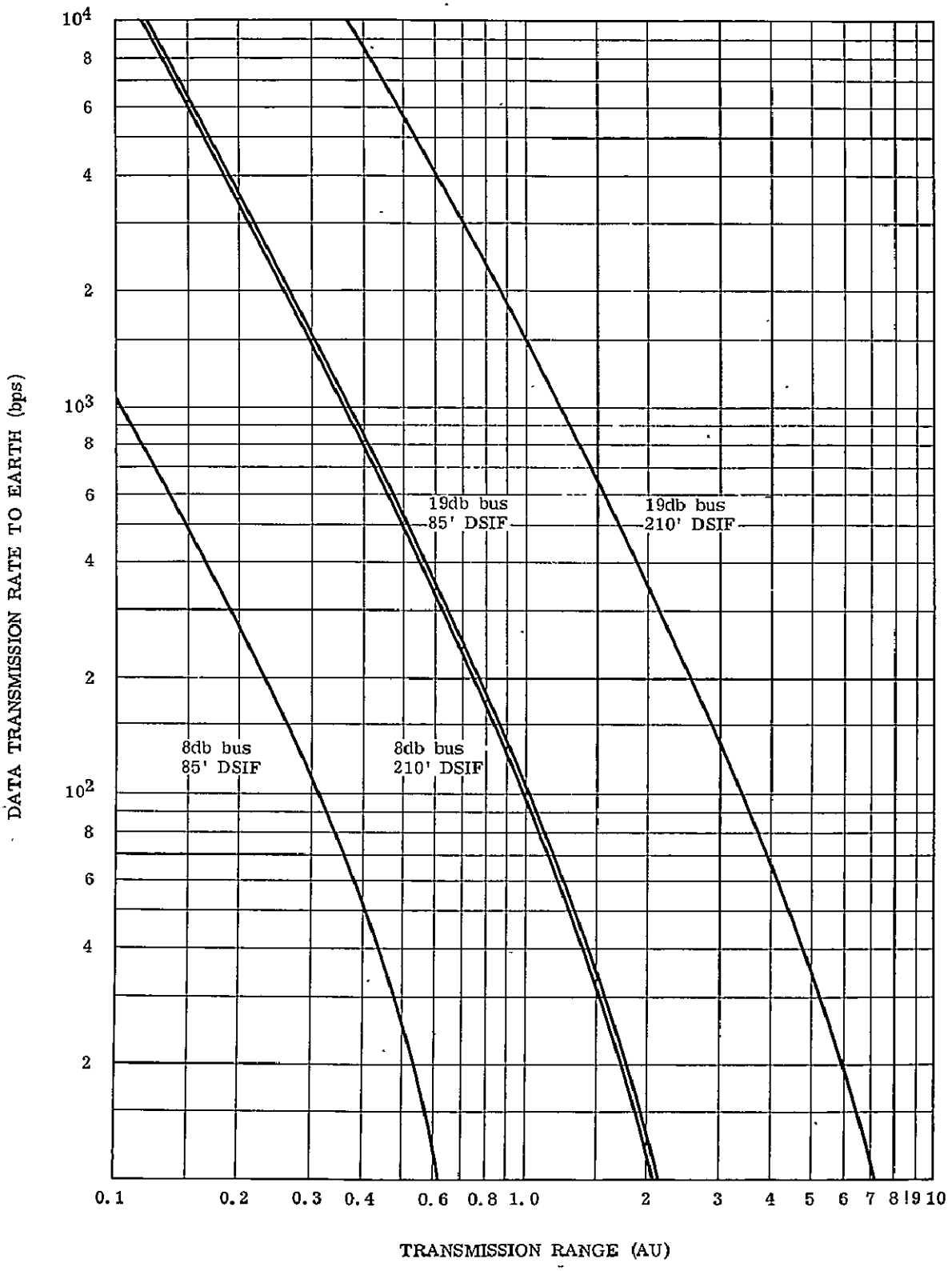


Figure 8. The curves are for the indicated bus and JPL Deep Space Instrumentation Facility (DSIF) antenna and for 10 watts of radiated power.

Command access to the orbiter bus will be available at all times through the use of the two boom-mounted omni-directional antennas shown in Figure 15.

2.5.4 Orbiter Thermal Protection

Cooling, when in the shadow of Venus, and additional heating due to Venus infra-red emission and to sunlight reflected from Venus are experienced by the orbiter. The extent of the problem depends upon inclination of the orbit plane to the Venus-Sun line and upon the periapsis and apoapsis distances. Solar cell temperatures of 110°C to -80°C occur for the orbiter which has a 1000 km periapsis over the subsolar point of Venus, and a 10,000 km apoapsis. A temperature change of less than 6°C occurs when the orbit plane contains the terminator circle and is thus at a right angle to the Venus-Sun line. Solar cell temperature extremes are found, however, to be tolerable and sufficient isolation of the instruments from the temperature extremes can be obtained.

2.5.5 Lifetime in Orbit

Lifetimes in orbit of 8 months are feasible.

2.6 BUS DEVELOPMENT STATUS

The GSFC Planetary Explorer Phase A report (Reference 1) and the AVCO study (Reference 2) indicate that the proposed bus vehicle can be built using available flight-proven or flight qualified systems, subsystems, and components. Although additional development of items such as the solid-state transmitter, the data-storage system, the course-correction system, and the electronically despun antenna would improve the bus performance and reliability, these systems exist and meet the bus requirements.

Additional contract study (Reference 3) is at this time directed to the improvement of the P. E. bus and to simplifying its conversion to the different requirements of the mission-series.

3. AERONOMY AND METEOROLOGY PROBE MISSION

3.1 MISSION SUMMARY

Ten days before Venus encounter the bus spin axis is reoriented to lie in the transit-orbit plane, the spin rate is increased to 100 rpm, and three small 52-pound entry probes are released tangentially, see Figure 9. The probes impact on Venus in the region of the sub-solar point, the anti-solar point, and the north pole point respectively, see Figure 10. During their 1.5 hour descent from 70 km to the surface, see Figure 11, the probes will be used to make measurements of atmospheric pressure, atmospheric temperature, and the Earth directed component of the horizontal wind.

Following small probe release, the bus is retargeted to the large probe entry location and the 380 pound large probe is released toward Venus.

Four hours before encounter and at a distance of 15 radii from Venus the bus science measurements will begin and will continue to an altitude of 130 km where communication with Earth is lost due to overturning of the bus as it enters the dense atmosphere. Bus science-instruments will measure the neutral and ion atmospheric-composition; electron-density and electron temperature; solar wind; and Venus magnetic field.

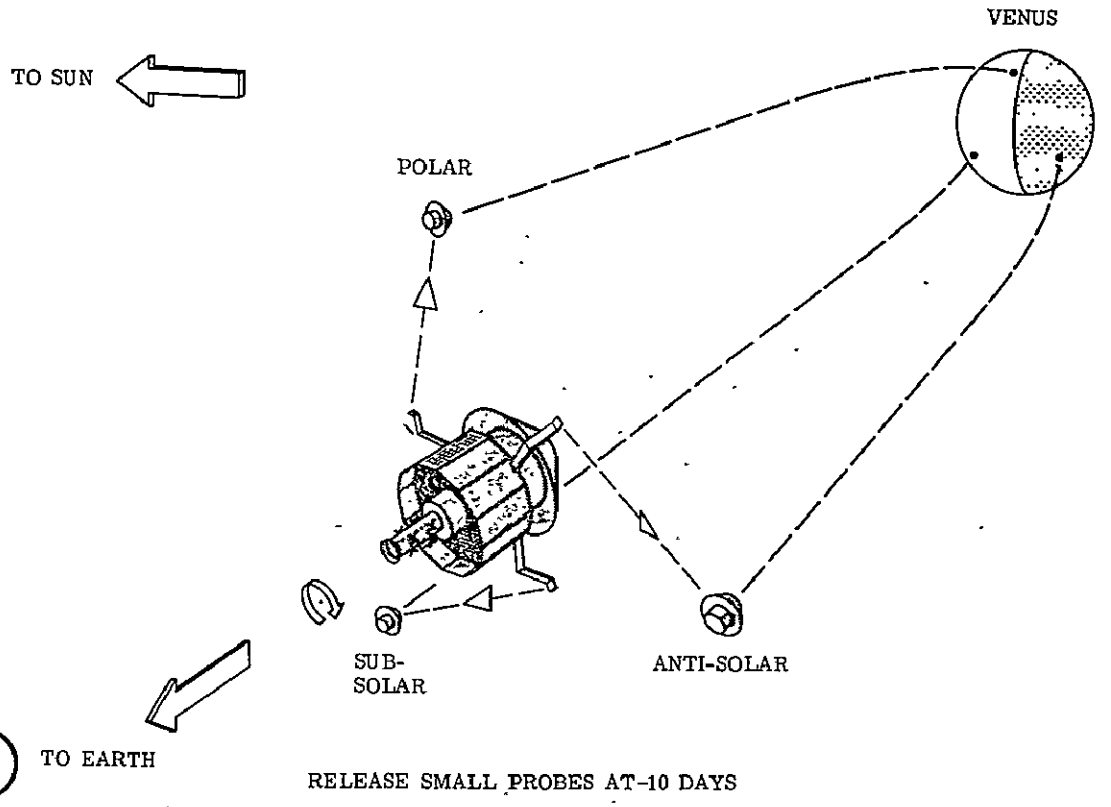
The large entry probe will make measurements in the region from 70 km to the surface. It will provide measurements of atmospheric pressure, temperature, and gas composition; solar radiation flux and upward thermal radiation flux; cloud particle composition, number-density, and size; horizontal wind speed; and altitude above the surface. The large probe is turned on by a timer 5 minutes before its deceleration by the Venus atmosphere. Deceleration measurements recorded by the large probe in the region 130 to 70 km will be used to compute the pressure, temperature, and density profiles needed to join the profiles of these same parameters obtained below 70 km to the profiles which will be obtained above 130 km by the bus measurements.

A summary of mission weights is given in Table 4.

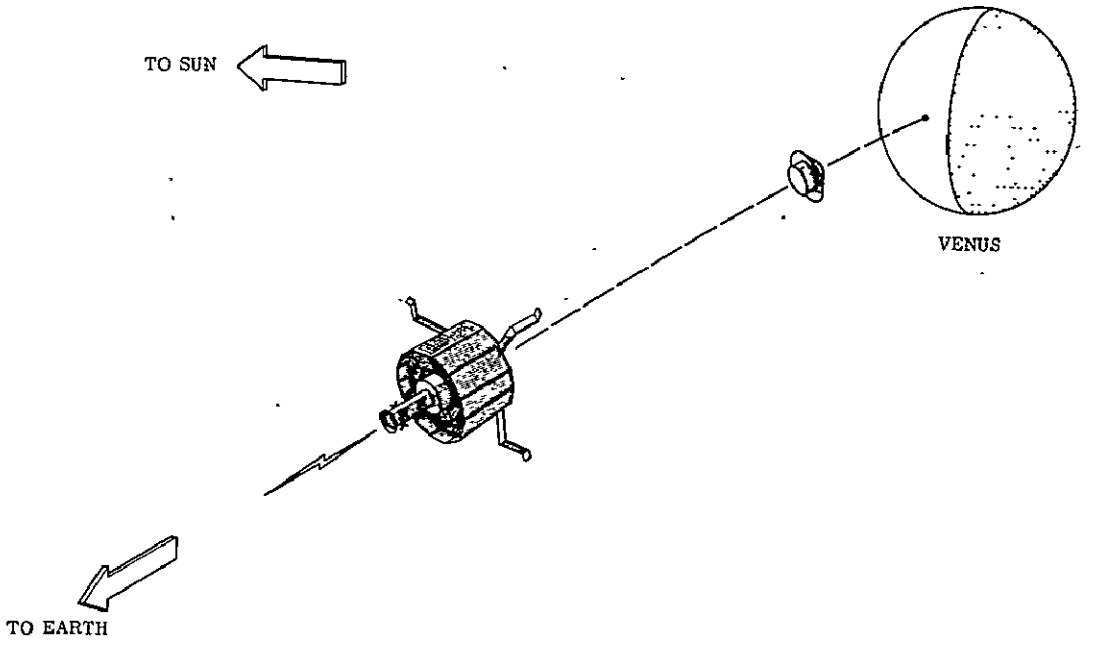
3.2 THE BUS

The bus and its use are described in Sections 2.1 and 3.1 and in Table 4. The bus follows the large probe on its path to Venus and enters at the same location, see Figure 10. The bus science measurements described in the preceeding section use instruments which have been flown on Earth satellites and space probes. No special problems arise in their use and no further instrument development is necessary.

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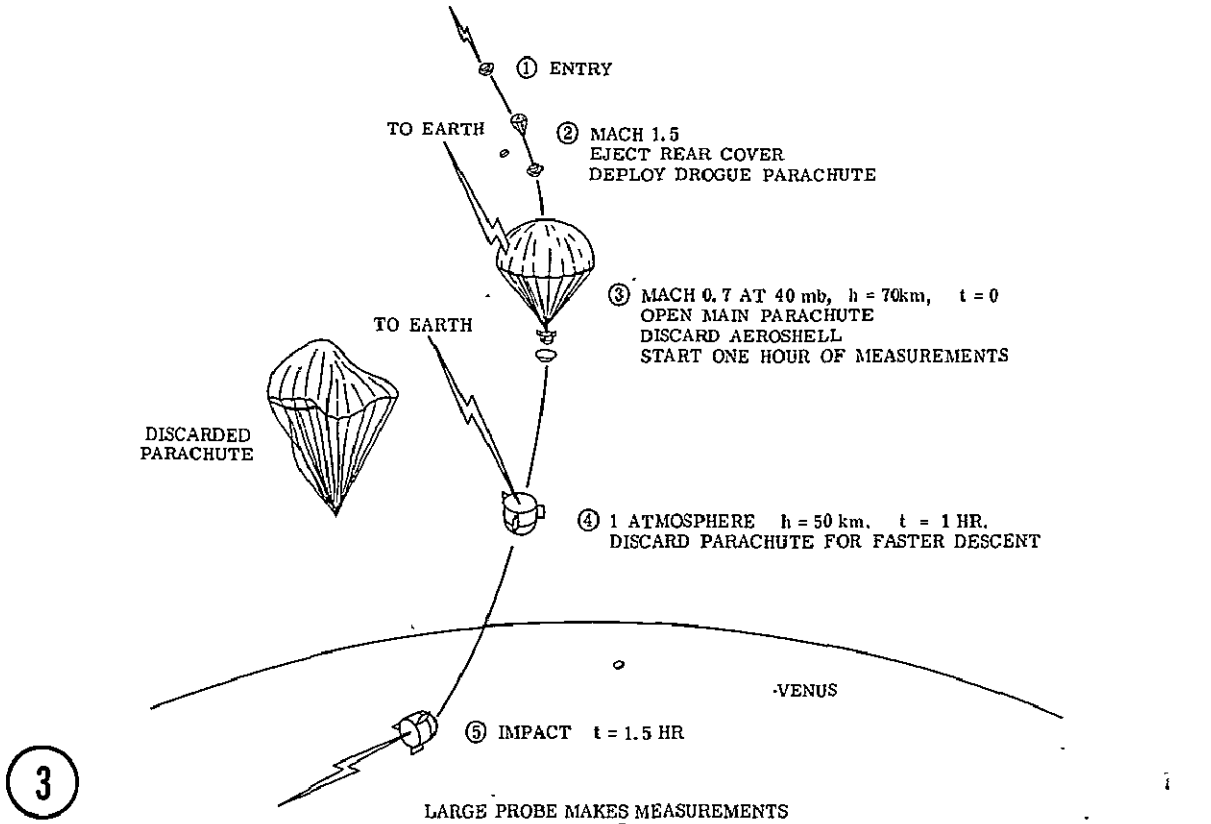


1 RELEASE SMALL PROBES AT -10 DAYS



2 RELEASE LARGE PROBE - START BUS MEASUREMENTS AT 15 RADII

Figure 9.



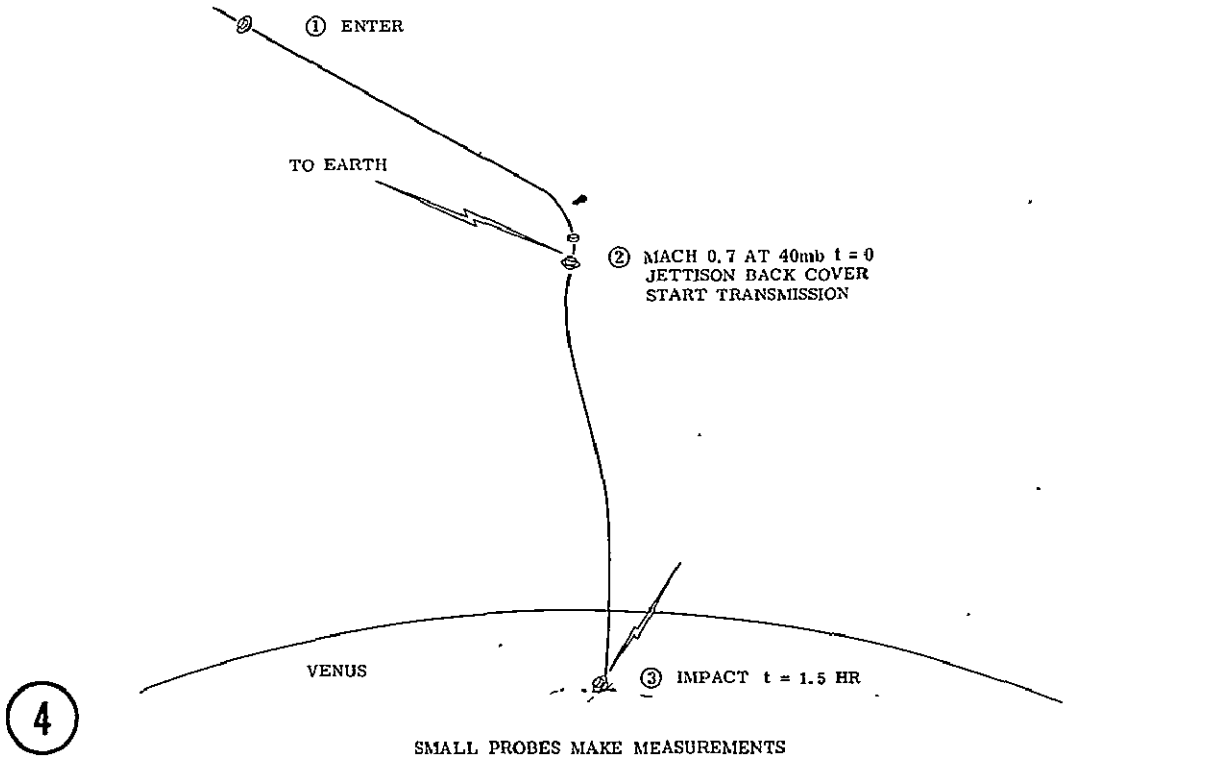


Figure 9. (Continued)

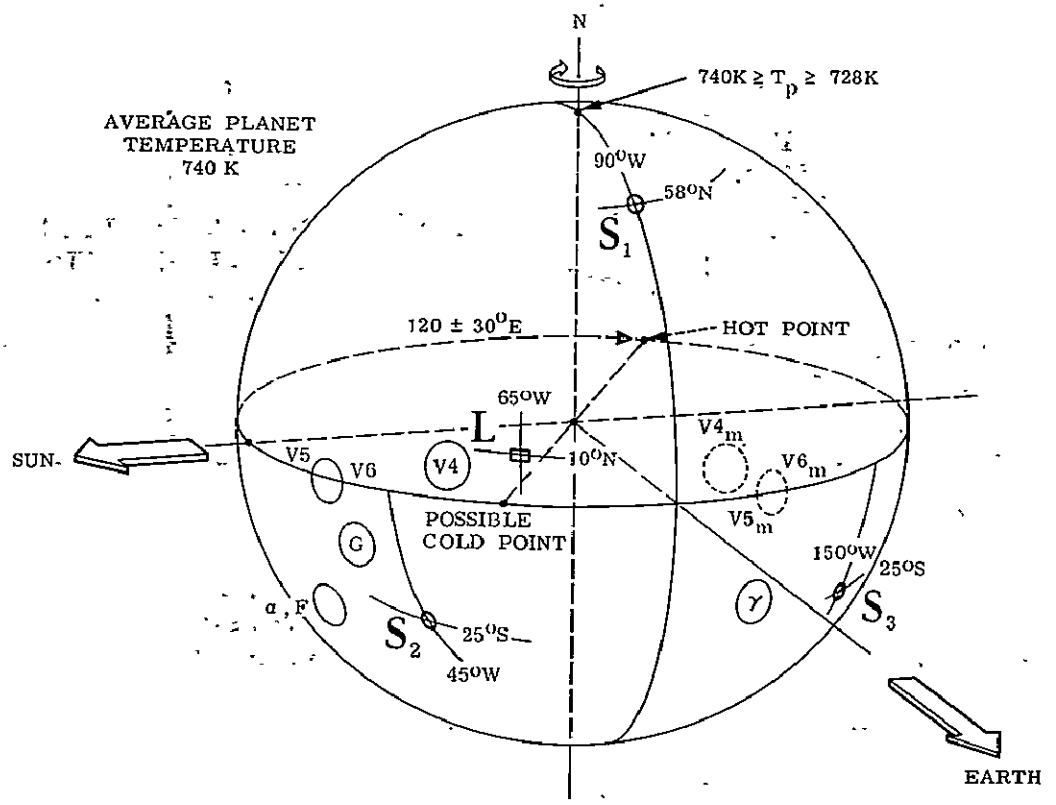


Figure 10. Small probe (S), large probe (L), and USSR Venera probe (V) landing points, and radar feature positions are shown. The Venera positions on the left are the points of impact on the surface and can be described in latitude and longitude with reference to the equator and radar feature α, F. The Venera positions on the right are given in solar or meteorological coordinates at the time of entry. Most of the radar features are on the far side of the planet. Temperatures were obtained by Sinclair, et al. (Reference 3). Maximum brightness temperature difference between the hot and cold points was found to be 18.4 ± 9.2K.

3.3 THREE SMALL PROBES

3.3.1 Purpose

In conjunction with the large probe, the three small probes determine whether diurnal variations exist in the altitude profiles of atmospheric temperature and whether the altitude profile of polar temperature differs from the profile obtained at the equator. Figure 10 shows recent planet surface temperature measurements and their relation to the small probe entry positions.

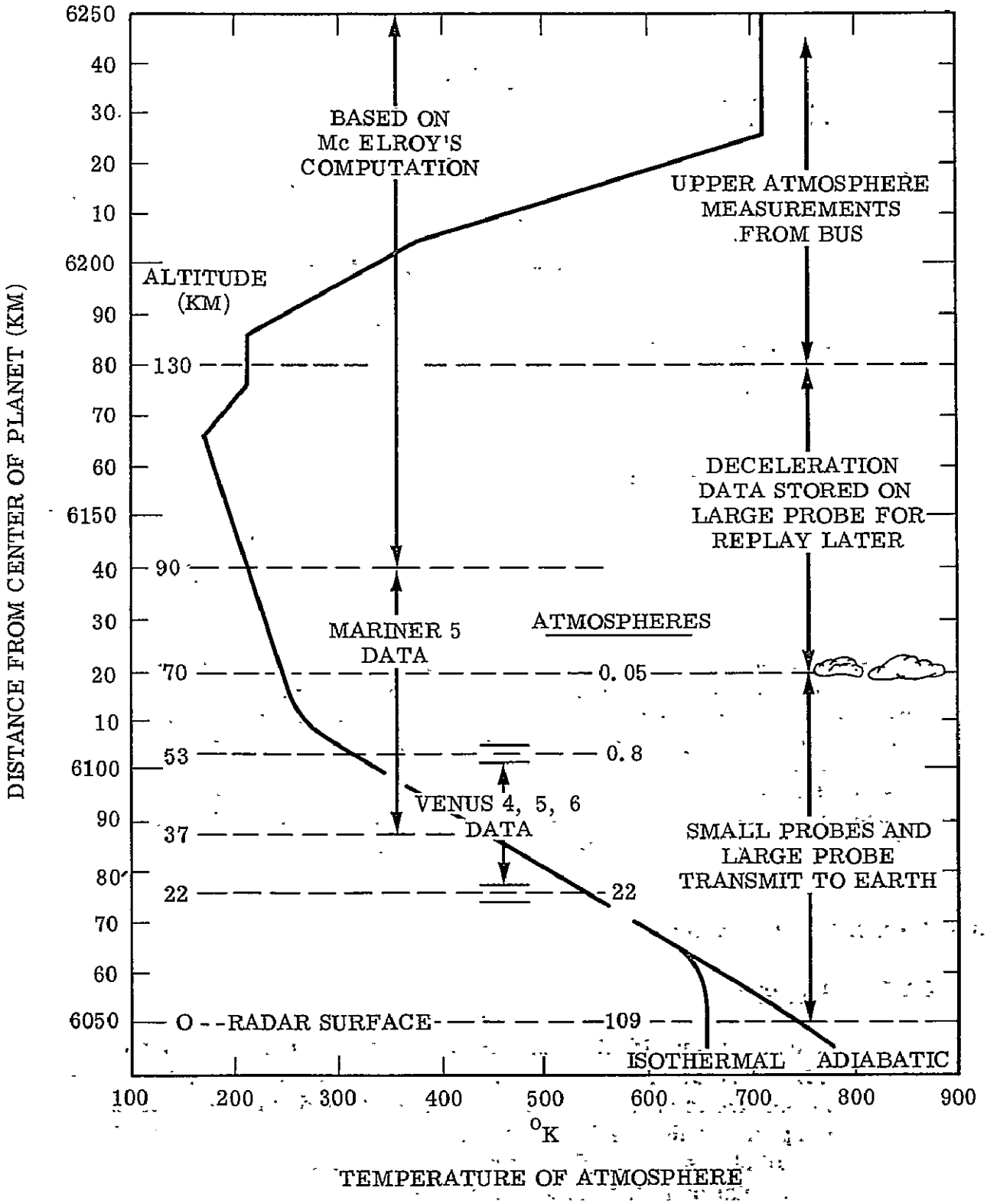

DISTANCE FROM CENTER OF PLANET (KM)
6250
40
30
20
10
6200
90
80
70
60
6150
40
30
20
10
6100
90
80
70
60
6050
ALTITUDE (KM)
130
90
70
53
37
22
0
BASED ON Mc ELROY'S COMPUTATION
MARINER 5 DATA
VENUS 4, 5, 6 DATA
O--RADAR SURFACE
UPPER ATMOSPHERE MEASUREMENTS FROM BUS
DECELERATION DATA STORED ON LARGE PROBE FOR REPLAY LATER
SMALL PROBES AND LARGE PROBE TRANSMIT TO EARTH
ATMOSPHERES
0.05
0.8
22
109
ISOTHERMAL
ADIABATIC
100
200
300
400
500
600
700
800
900
°K
TEMPERATURE OF ATMOSPHERE

Table 4

Mission Weight Summary

Item	Pounds	
Bus science	25	
Bus total		304
Large probe science	70	
Large probe total		380
Small probe science 3 × 4	12	
Small probe total 3 × 52		156
Total science weight	106	
Total system weight		840
1975 Delta launch capability		844

The small probe temperature measurements and the measurement of their Earth-directed component of wind drift will assist in defining an atmospheric circulation model.

3.3.2 Probe Release

Immediately before small probe release, the bus spin-rate is adjusted so that the tangential velocities of the small probes at the time of their release will take them to their required Venus impact points. Proper targeting of the small probes can be achieved by commands to the bus which (a) change its spin rate, (b) change the phase of its rotation with respect to the sun at the instant of release, and (c) change the number of days before entry at which release occurs. The errors in release are ±1° in azimuth angle, 0.6% in speed, and 10^{-3} sec. in time. Uncertainty in the position of the bus at the time of small probe release is 100 km.

The rotation rate of a small probe remains unchanged by separation. Since atmospheric entry requires a low spin rate in order to assure rapid angle of attack reduction, the probes are de-spun to 15 rpm by means of a yo-yo just after release from the bus.

3.3.3 Landing Points

All three small probes transmit directly to Earth and it is the combination of attenuation and defocusing of the radiated signal by the dense Venus atmosphere at low Earth elevation angles which determines how closely the probes can be landed to the desired subsolar, anti-solar, and polar points. The two effects establish the requirement of a nominal beam elevation of 25°, or more, above the Venus surface. Probe skip-out, which occurs for entry angles* less than 14°, is not a problem since gravitational bending of the trajectory insures entry angles of from 25 to 80° under all conditions where the 25° communications requirement is met.

3.3.4 Probe Configuration

The configuration is shown in Figure 12. Probe hypersonic and subsonic ballistic coefficients, $m/(C_d A)$, are 0.8 and 1.1 slugs/ft.^{2}, respectively.

Pressure and temperature sensitive components of the probe instruments are enclosed in a thick-wall aluminum pressure vessel capable of withstanding an external pressure of 200 atmospheres (3000 psi). The pressure vessel is filled with sulfur hexafluoride at 15 psi in order to inhibit voltage breakdown and to allow convective heat transfer to assist in preventing instrument hot spots. Sodium sulfate dekahydrate, which changes from a crystalline solid to a liquid at 32.8°C, is used within the pressure vessel to absorb both heat generated by the probe instruments and heat entering the vessel from outside. The pressure vessel is attached to the aeroshell by a thermal isolation ring and is surrounded by a blanket of high temperature, high pressure thermal insulation.† Release of the back cover deploys the temperature sensors. Small probe weight breakdown is given in Table 5.

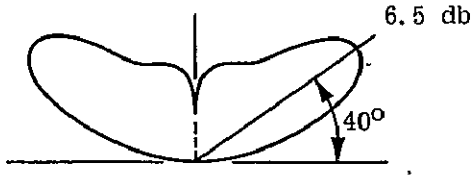
3.3.5 Communication

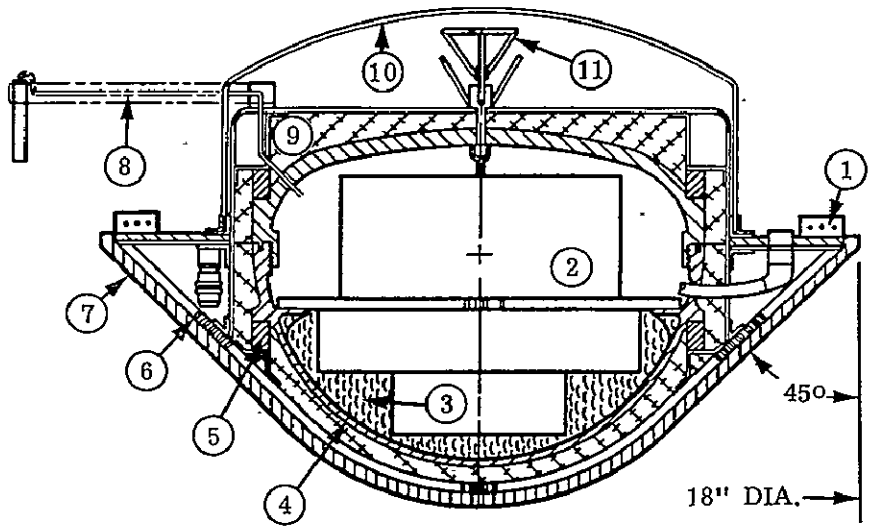
Communication is direct to Earth by means of a 2300 MHz S band solid-state transmitter radiating 2 watts and transmitting at 1 bps to the DSN 210 ft. antennas.

*The path angle above the surface at entry.

†Min-K(TE) consists of submicron silica particles, an opacifier to minimize radiation transfer, and silica fibers for reinforcement. It has the lowest thermal conductivity of any non-vacuum insulation, has low outgassing, low linear shrinkage, is load bearing,and may be moulded and machined.

COMMUNICATION FIELD INTENSITY





1 SURFACE APPROACH INDICATOR ANTENNA FEED
2 TIMER, TRANSPONDER, TEMPERATURE GAUGE ELECTRONICS, PRESSURE GAUGES (3), BATTERY, IMPACT INDICATOR ELECTRONICS
3 PHASE CHANGE MATERIAL
4 PRESSURE VESSEL
5 STRUCTURAL INSULATION RING
6 AEROSHELL - TITANIUM HONEYCOMB
7 HEAT SHIELD-CARBON PHENOLIC
8 TEMPERATURE GAUGE (DEPLOYED)
9 THERMAL INSULATION
10 REAR COVER
11 COMMUNICATION LOOP-VEE ANTENNA

Figure 12. Small probe configuration.

Transmission starts 5 minutes prior to entry blackout in order to assess whether all instruments are operating properly and to allow adequate warm-up.

Table 5

Small Probe Weight-Breakdown

Item	Pounds
Science instruments	3.8
Communication system	5.0
Power system	14.7
Pressure vessel, heat sink, and insulation	11.9
Heat shield and structure	12.4
Contingency	4.4
Total weight on bus:	52.2

3.3.6 Measurement

The small-probe science-instruments comprise:

a. A temperature gauge with two ranges, 200-500 K and 500-800 K, and automatic range switching.

b. Three pressure gauges with ranges of 0 to 1.5, 0 to 15, and 0 to 170 atmospheres. The output of a single gauge only is transmitted and gauge switching proceeds automatically.

c. A surface approach indicator with a range of 200 meters. The indicator, which excites the aeroshell for use as an antenna, will inject at least three consecutive approach indication signals into the data stream just before the small probe impacts. Loss of the small probe signal will then be known to have been due to impact on the surface rather than due to probe failure.

Approximately 250 temperature measurements and 250 pressure measurements will be obtained during small probe descent. The interval between successive pressure measurements will be about 1.5 km at 70 km and 0.05 km at the surface. All data will be encoded in 8 bit words.

3.3.7 Winds

If large vertical winds are absent at the small probe descent positions, a vertical profile of the Earth-directed-component of the horizontal drift speed of the descending probe can be obtained from the Doppler shift of the probe transmitter frequency. This single-wind-component profile can be of value in selecting among possible circulation models. In Figure 13(a) the Earth-directed-component of the wind speed obtained for the sub-solar and anti-solar probes has a maximum horizontal-component roughly equal to the true horizontal-wind velocity, while the wind-velocity component obtained from the polar probe is small. In Figure 13(b) the situation is reversed with small indications from the sub- and anti-solar probes and a polar-probe Earth-directed wind-component which has a maximum horizontal-component roughly equal to the true horizontal wind-velocity.

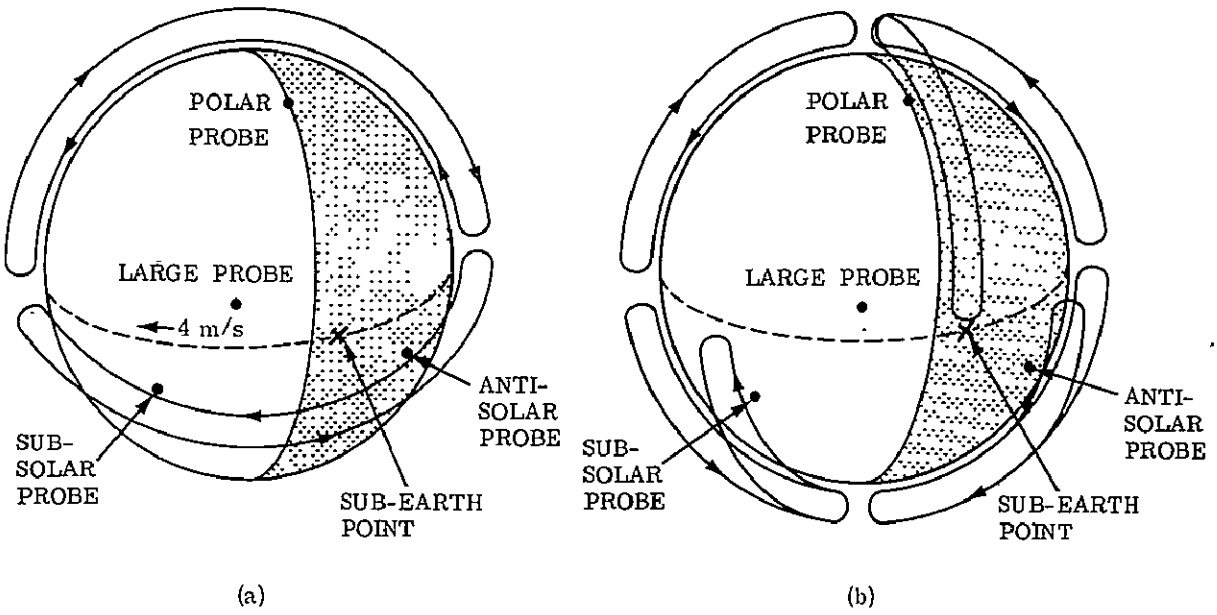


Figure 13. Two possible atmospheric circulation models. The models are used for illustration; planet rotation and the pattern of solar energy penetration may lead to a more complex circulation pattern. The Venus surface moves under the terminator at 4 m/s, or 3.1° per day.

The accuracy with which the Earth-directed component of the horizontal wind can be measured depends upon:

a. The error in the measured probe Doppler-shift. If a turn-around transponder is used on the probe, the error is small. For the small probes, which use a crystal controlled transmitter, Doppler error is considerably larger due to frequency drift.

b. The error in our knowledge of the Earth-Venus departure speed at the time of measurement; in the rotational departure speed at the probe entry position; and in the rotation departure speed at the Earth radar location. The error in the rotational departure speeds is negligible.

c. The speed of any vertical winds. For the small probes, vertical winds which are less than 0.5% of the probe descent speed cause a negligible error.

d. The error in our knowledge of the probe descent speed. This error is large and arises mainly from uncertainty in the computed atmospheric density and the uncertainty in the ballistic coefficient of the probe following the loss of heat shield material by ablating during entry.

Table 6 shows the resulting total error that would be obtained in the use of the Doppler signal from a small probe to determine the speed of a horizontal wind directed along a great circle through the sub-Earth point.

Table 6

Small Probe Wind Speed Error*

Altitude (km)	Descent Speed m/s	Total Wind Speed Error (m/s)
70	176	5.6
50	38	2.4
30	15.5	1.9
0	6	1.6

*Total error obtained in the use of the Doppler signal from a small probe transmitter to determine the speed of a horizontal wind directed along a great circle through the sub-Earth point. It is assumed that no vertical winds exist.

If, as has been suggested, a 4-day high-altitude retrograde wind exists, conversion of the probe Doppler measurement to an equatorial wind at 70 km altitude would yield a wind speed of 110 ± 6 m/s. The measurements will confirm or refute the existence of winds of this speed along equator.

Short-period changes in Doppler shift are independent of transmitter frequency drift and can be measured more accurately than the absolute Doppler shift. Thus, the Earth-directed speed-component of turbulence induced motions of the small probe can be measured to within ±40 cm/sec.

3.4 THE LARGE PROBE

3.4.1 Purpose

The large probe is designed to accomplish the principal mission objectives. It examines the Venus cloud cover to discover (a) the degree to which it is penetrated by solar flux from above and by thermal flux from below, (b) the variation in size and number density of its particles with altitude, and (c) the materials of which it is composed. Other uses include obtaining detailed altitude-profiles of atmospheric temperature and pressure starting from the surface; obtaining more precise measurement of atmospheric gas-composition and searching for trace constituents; measurement of horizontal wind speed during descent; and determining the scale of surface physical features in the vicinity of the impact point. In conjunction with the small probe measurements, large probe measurements will assist in defining the diurnal temperature variation, the polar-equatorial temperature variation, and the atmospheric circulation pattern.

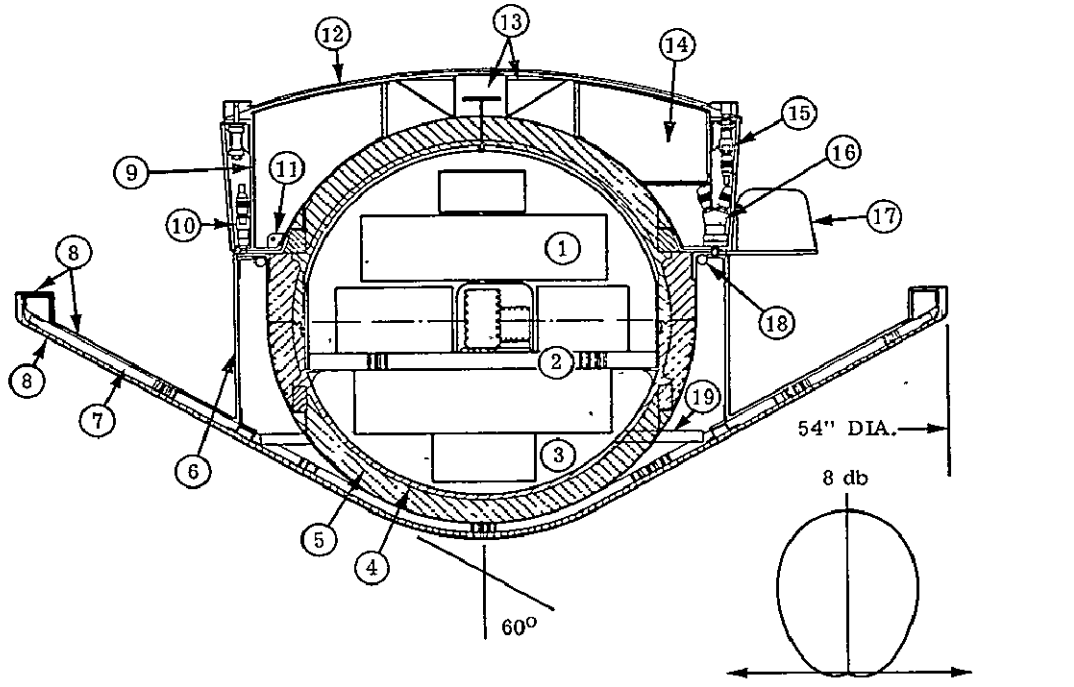
3.4.2 Separation From the Bus

Following small probe release the bus is retargeted and despun; the explosive bolts holding the large probe to the bus are fractured, and springs force a separation of the probe from the bus at 1 meter/sec.

3.4.3 Entry and Descent

During entry black-out there is on-board storage of probe deceleration measurements for replay during descent. Parachute deployment is shown in Figure 9. Discarding of the aeroshell permits the

use of downward-looking optical and radar measurements. The slow descent obtained by use of a parachute permits extensive measurements in the region from 40 mb to one atmosphere. The time that can be spent in this region is determined by parachute weight, internal heating, and battery weight considerations. At lower altitudes, however, the high ambient atmospheric temperature requires a faster descent and the parachute is released. Descent speeds are given in Table 9. Fins, see Figure 14, are attached to the probe in order to obtain the probe rotation required by the solar radiometer at high altitudes and by the drift radar at lower altitudes. To permit rotation, the main parachute is attached to the probe by means of a swivel.



1 INSTRUMENTATION AND ELECTRONICS
2 EQUIPMENT SHELF
3 PHASE CHANGE MATERIAL
4 PRESSURE VESSEL
5 THERMAL INSULATION
6 SUPPORT STRUCTURE
7 AEROSHELL
8 HEATSHIELD
9 PARACHUTE CANISTER
10 PARACHUTE CANISTER RELEASE
11 PARACHUTE TIEDOWN
12 REAR COVER AND HEATSHIELD
13 ANTENNA-CIRCULAR APERTURE
14 MAIN PARACHUTE
15 REAR COVER EJECTOR
16 AEROSHELL EJECTOR
17 SPIN FIN 3 REQD
18 RADAR ALTIMETER ANTENNA
19 CLOUD PARTICLE COUNTER

Figure 14. Large probe configuration.

3.4.4 Large Probe Configuration

The configuration is shown in Figure 14. Probe hypersonic ballistic coefficient is 0.5 slugs/ft.^{2}. Ballistic coefficient with the main parachute attached is 0.01 slugs/ft.^{2}. When the main parachute is jettisoned at 50 km the ballistic coefficient of the sphere-cylinder descent vehicle is 8 slugs/ft.^{2}.

The structure of the pressure vessel is similar to that of the small probe pressure vessel described in Section 3.3.4. Large probe weight-breakdown is given in Table 7.

Table 7

Large Probe Weight-Breakdown

Item	Pounds
Science	70
Transponder and antenna	11
Power and data handling	40
Pressure vessel	55
Insulation and heat sink material	28
Aeroshell and miscellaneous structure	62
Heat shields (fore and aft)	46
Separation hardward	6
Parachute (40 ft. dia.) and mortar	35
Contingency	27
Total weight on bus	380

3.4.5 Communication

Large probe communication is direct to Earth by means of a 2300 MHz S-band solid-state turn-around transponder radiation 10 watts and uses the DSN 210 ft. antennas. Transmission starts at 80 bps 5 minutes prior to entry black-out in order to assess whether all instruments

are operating properly and to allow adequate warm-up. Following parachute release at 50 km, the transmission rate is reduced to 40 bps.

3.4.6 Measurements

3.4.6.1 Acceleration. The accelerometers measure probe deceleration in the region from 130 to 70 km and the measurements are stored for replay during the latter part of parachute descent. The acceleration measurements are used with measurements of the mean-molecular-mass of the atmosphere, the computed altitude versus time profile, and the computed probe ballistic coefficient to construct the profiles of atmospheric pressure, temperature, and density. This information bridges the gap between 130 km where bus measurement of upper atmosphere parameters ceases and 70 km where lower altitude measurements start during parachute descent.

3.4.6.2 Pressure. The five pressure gauges used will have ranges of 0.2, 0.8, 4, 25, and 170 atmospheres. In order to reduce the amount of information required to be transmitted, pressure actuated range switching will be used. Measurement error before telemetry will range from 1/2 to 1% of full scale for each gauge. The altitude interval between pressure measurements is given in Table 8.

3.4.6.3 Temperature. A single temperature sensor will be used with temperature switched ranges of 200-300, 300-400, 400-600, and 600-900°K. Measurement error before telemetry will be ±0.5% of the temperature range or ±0.5° for the two low temperature ranges and ±1.5° for the high temperature range.

3.4.6.4 Atmospheric Gas-Composition. The mass spectrometer inlet comprises an inlet break-off system, a calibrated leak leading to a ballast volume and a second calibrated leak leading from the ballast volume to the mass spectromemter ionizing region. The principal constituents of the atmosphere will be more accurately defined and a search will be made for trace constituents in the range 1 to 90 AMU.

3.4.6.5 Water Vapor. The sensing element is a water vapor sensitive capacitor comprising a $10 \times 5 \times 0.1$ mm strip of anodized aluminum coated with a thin vacuum-deposited layer of gold. The gold layer is porous and allows rapid diffusion of water vapor in and out of the aluminum oxide with a resulting response time of a few seconds throughout the range of the instrument. The instrument is capable of measuring dew or frost points at -110 to +20°C (164-294°K), corresponding to water vapor pressures of 10 to 10^{-5} mb or water vapor densities of 20,000 to .001 microgram/liter of water.

Table 8

Typical Sampling Intervals. The intervals are for a descent speed of 15 m/s and a transmission rate of 80 bps and can be converted linearly for other descent speeds (see Table 9) and for 40 bps transmission.

Parameter	Sample Interval (km)
Pressure	1
Temperature	1
Atmospheric gas composition (1-90 AMU)	1
Water vapor (hygrometer)	0.3
Thermal radiation (upward flux)	0.1
Solar radiation (4π steradians and 4 wavelengths)	1
Cloud backscatter (nephelometer)	0.1
Cloud particle number-density and size	0.1
Cloud particle composition (3-300 AMU)	7.5
Condensation/evaporation	1.6
Altitude (radar)	0.5

Recent experiments indicate no temperature dependence from -110 to +85°C (164-359°K). Nor is there dependence upon total ambient pressure over a range of 10^{-6} to 100 atmospheres. Maximum error is ±2° in indicated dew/frost points. The sensor is unaffected by the presence of a variety of other gases, and is also capable of measuring the water vapor pressure in liquid hydrocarbons as well.

The range of operation on Venus is expected to be from 40 to 100 km altitude (6090 to 6150 km radius).

3.4.6.6 Thermal Radiation. The thermal radiometer is mounted looking downward. In a near adiabatic atmosphere the difference between the brightness temperature measured in a CO_2 window at 7μ and a CO_2 absorption region at 10.4μ is proportional to the transmission

characteristics of the clouds. Small differences correspond to opaque clouds, large differences to thin or absent clouds. The instrument thus provides a vertical profile of effective cloud density and will detect whether stratified condensable clouds exist or whether there is a continuously distributed dust cloud.

The experiment provides a remote surface temperature measurement just before probe impact.

3.4.6.7 Solar Radiation. The field-of-view of the radiometer is divided into five 30° conical sections starting at 20° from the zenith and extending to 10° from the nadir. Probe rotation results in almost spherical coverage. Beam splitting allows the separation of each of the 5 light inputs into 4 different wavelength ranges for a total of 20 different input signals. These 20 measurements are made every 60° of probe revolution or 6 times per revolution.

The divergence of the solar flux which is derived from these measurements represents the fundamental dynamic drive of the atmosphere.

3.4.6.8 Cloud Back-Scatter (Nephelometer). The nephelometer measurement of scattered light gives an indication of cloud density and reflectivity. Measurement during descent will indicate whether the cloud cover is a stratified condensable or is a continuous dust cloud.

3.4.6.9 Cloud Particle Number Density and Size. Individual cloud particles will be both counted and measured by optical means in the size range 1 to 250 microns and all particles greater than 250 microns in size will be counted. The instrument, now under development, observes the shadows of the particles against a fiber-optic detecting mosaic, a method which is tolerant of background light.

3.4.6.10 Cloud Particle Composition. In addition to the gas-source mass spectrometer, which is also to be used to analyze evaporated condensate from the cloud particles, the large probe must carry a solid-source mass spectrometer to analyze cloud particle nuclei and dust cloud particles which may consist of salts or mineral materials from the surface. It would be desirable to measure masses in the range of 3 to 300 atomic mass units. Preliminary studies must be concerned with developing techniques for collecting, evacuating, and evaporating the sample. The particle collection system must comprise a collector tube, filters, high temperature stepping units, high temperature solenoid valves, and ballast volumes. Prior to its presentation to the mass spectrometer the

sample will be placed in a chamber in which condensate material will be evaporated. The gases from this evaporation will be analyzed in the gas mass spectrometer. Presently, untreated rock minerals are analyzed using spark-source mass spectrometry. Modification of bulky spark generating equipment to a suitable size and power rating may be possible. An alternate method is the use of a laser beam to evaporate the sample. The presence of sample material in the laser beam could be determined by measuring light scattered from the sample.

Sample vaporization would take place outside the pressure vessel and in order to minimize loss of sample through condensation, the sample injection tube must allow a straight-line path from the vaporized sample to the ionization region of the mass spectrometer located in the pressure vessel.

A possible alternative to the use of mass spectrometry is the bombardment of the sample with electrons and the identification of the sample by detection of X-ray fluorescence or Auger electron emission.

3.4.6.11 Condensation-Evaporation. A typical instrument comprises a light source and a small reflector external to the pressure vessel. The image of the source is reflected into the sphere where it is observed by a photocell. The small reflector is cycled in temperature every 300 seconds from 25°K below to 25°K above ambient temperature, and then to 25°K below to start the cycle again. Atmospheric gas is dust-filtered and directed at the reflector. The photocell observes changes in the reflected light resulting from:

a. "Dew" forming on the mirror. The temperature of the mirror at which a liquid forms is compared with the ambient atmospheric temperature to determine the percent of saturation of the atmosphere by the liquids' vapor.

b. "Frost" formation. The temperature at which frost occurs is either the sublimation temperature or the melting temperature of the material. Sublimation temperatures can be identified by the absence of a liquid phase. Measurement of the melting temperature assists in the identification of the condensate since there are a limited number of plausable substances with melting points lying within the uncertainty of the melting-temperature-measurement.

3.4.6.12 Altitude and Drift Radar. The altitude radar is a conventional S-band FM unit with a downward looking antenna. Altitude range is 6 km. Altitude measurement error will be ±100 meters or less before telemetering. Descent speed error will be ±10 cm/s or less. Counting techniques can be employed if smaller error in range and descent speed are desired. For various combinations of horizontal wind speed and surface feature dimensions, a crude description of the physical features of the surface may be possible by combining the altitude and drift radar measurements.

The drift radar is a conventional S-band FM unit mounted within the large sphere and with transmitting and receiving antennas looking 30° down from the horizon. Maximum range of the unit will be 6 km. Error before telemetering will be ±100 meters or less. Probe rotation rate near the surface is 15°/sec. and each measurement will include a 4 second or 60° count of the Doppler frequency difference. The use of counting will permit measurement of drift speeds as low as 2 cm/s with an error of ±1 cm/s. Maximum drift speed measurement will be set at 2 m/s.

Range measurements by the drift radar may enable the identification of surface physical features.

3.4.6.13 Carrier Transponder. Large probe speed-measurement during descent utilizes a carrier transponder. In the absence of horizontal winds on Venus this instrument will permit Doppler measurement from Earth of probe descent speed to within ±3 cm/s. Thus turbulence or vertical wind changes of this magnitude will be measurable. The error in the use of the transponder to determine the true vertical wind-speed is for the most part determined by the errors in the measured atmospheric density and in the post-flight knowledge of the probe's ballistic coefficient. The resulting error is roughly ±2.5% of the probe's descent speed. From Table 9 we see that at 50 km the error in the vertical wind speed measurement would be approximately ±10 cm/s before parachute release.

In the absence of vertical winds, the error in the measurement of the horizontal speed of the large probe along a great circle through the sub-radar point is given in Table 9. Vertical winds which are less than 0.5% of the probe descent speed cause a negligible change in the speed errors given in Table 9. Thus, for winds greater than the largest of the error values given in Tables 6 and 9, components of the horizontal wind speed over the altitude range 0-70 km will be measured by 4 probes (one large, three small) at 4 different positions on the planet.

Table 9

Large Probe Wind Speed Error*

Altitude (km)	Descent Speed (m/s)	Total Wind Speed Error (m/s)
70	18	±0.5
50	4	0.2
50	105	2.8
30	42	1.1
0	17	0.5

*Total error obtained in the use of the Doppler signal from the large probe turn-around transponder to determine the speed component of a horizontal wind along a great circle through the sub-Earth point. It is assumed that no vertical winds exist. The parachute is deployed at 70 km and subsequently released at 50 km.

3.5 MISSION STATUS

Preliminary studies by GSFC, an advanced study by AVCO (Reference 2), detailed review of the AVCO study by the GSFC Planetary Explorer study team, additional work by AVCO (Reference 3) and related work by Martin-Marietta for JPL (Reference 5) have resulted in a firmly based knowledge of the capability of the aeronomy and meteorology probe mission.

Of the 17 kinds of science instruments carried on the mission, see Table 10, 11 can be used essentially as they have been used on past rocket, satellite, and space missions. Of the remaining 6 instruments the cloud particle number-density and size detection instrument is under development; the gas composition mass spectrometer requires the manufacture and testing of a 170 atmosphere inlet system; the particle composition mass spectrometer requires the development of a sample introduction system; the thermal and solar radiometers, and the condensimeter-evaporimeter remain to be developed. The remaining instrument development, however, is based on long established methods and will be able to be completed by 1973 for a 1975 mission.

Table 10

Summary of Aeronomy and Meteorology Mission Instruments

Vehicle	Instrument	Weight (lbs.)	Power (watts)	Volume (in^3)
P.E. bus	Neutral Particle mass spectrometer 1-50 AMU	8.0	10.0	500
	Ion mass spectrometer	3.0	1.0	330
	Electron temperature probe	2.0	1.5	90
	Solar-wind plasma probe	7.0	5.0	200
	Magnetometer	5.0	1.0	130
	Totals	25.0	18.5	1250
Small probe	Temperature gauge (2 sensors)	1.2	0.15	60
	Pressure gauges (3)	0.8	0.05	10
	Surface approach indicator	2.0	3.00	100
	Totals	4.0	3.20	170
Large probe	Accelerometers (4)	2.0	4.0	20
	Pressure gauges (5)	1.3	0.1	10
	Temperature gauge	1.2	0.15	60
	Neutral particle mass spectrometer 1-90 AMU	10.0	12.0	500
	Hygrometer	1.0	3.0	100
	Thermal radiometer	3.0	3.0	20
	Solar radiometer	4.0	3.0	100
	Nephelometer	4.0	3.0	100
	Cloud particle number and size	4.0	10.0	190
	Cloud particle composition 3-300 AMU	20.0	60.0	500
	Condensimeter/evaporimeter	2.0	10.0	70
	Radar altimeter (10 km)	4.5	18.0	100
	Wind drift radar (10 km)	4.5	18.0	100
	Turn-around transponder	8.5	35.0	100
	Totals	70.0	179.25	1970

4. FIELDS, PARTICLES, AND AERONOMY ORBITER MISSION

4.1 MISSION SUMMARY

The orbiter shown in Figures 7 and 15 is placed in the orbit shown in Figure 16 with the instruments of Table 11. Mission objectives are to:

a. measure the detailed structure of the ionosphere and upper-atmosphere of Venus in-situ,

b. measure the interaction of the solar wind with the ionosphere of Venus,

c. determine the presence or absence of an intrinsic magnetic field for Venus and investigate the implications in regard to its internal composition,

d. determine the presence or absence of a magnetic bow shock and tail, or wake, for Venus and compare with other planetary environments in the solar system,

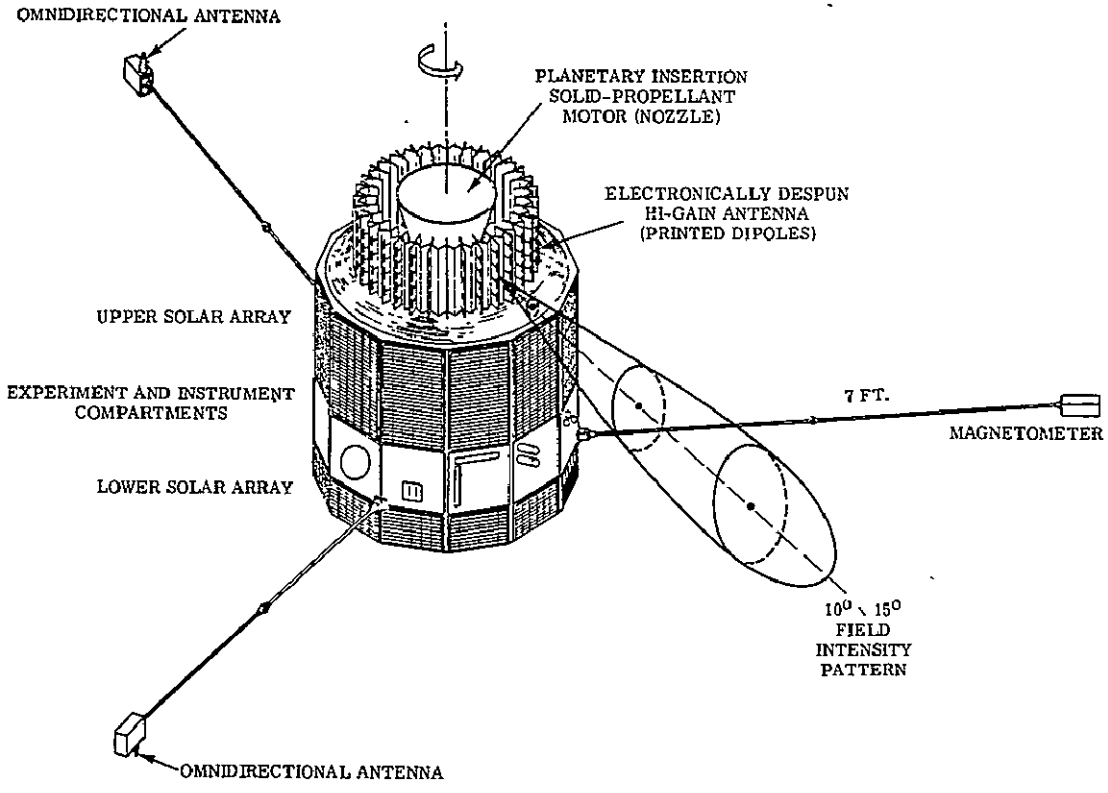


Figure 15. Fields, particles, and aeronomy orbiter.

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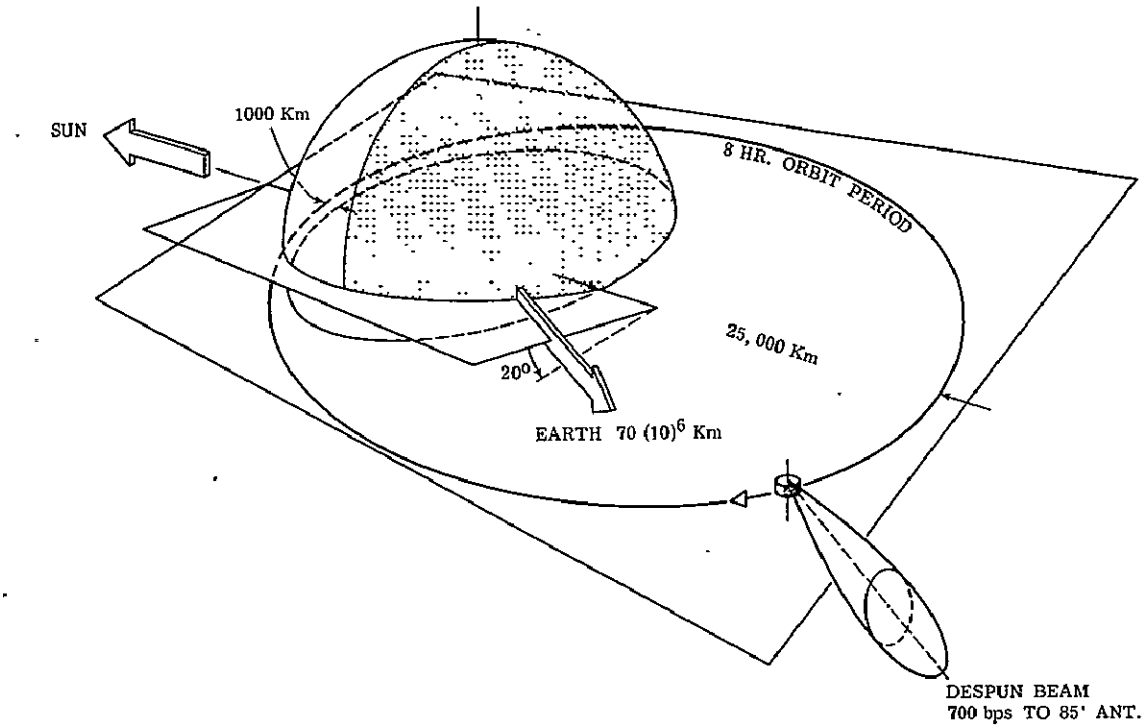


Figure 16. The orbit for the fields, particles, and aeronomy mission arriving approximately May, 15, 1977. A 1500 km periapsis change can be obtained when in orbit. Data storage is used when Earth is occulted by Venus.

e. measure the interplanetary magnetic field and solar-wind plasma en-route to and in the vicinity of Venus, and study spacecraft orbital motion to determine possible Venus gravitational field harmonics.

4.2 MEASUREMENT

For a given selection of science instruments carried by the orbiting bus, the interval between consecutive measurements of the same parameter depends upon the instrument measuring speed and the data storage capability. If data transmission is in real time, the interval depends upon instrument measuring speed, the transmitter power, the transmission distances, and the bus and DSIF antenna gains. For the 1976-77 mission described here, real time transmission by the despun bus antenna to the 85' DSIF antenna, see Figure 8, allows a 2 second sampling interval. With the use of the 210' antenna an 0.2 second sample interval is obtained.

4.3 MISSION VARIATIONS

The removal of the 35 pounds of hydrazine fuel necessary to achieve the in-orbit periapsis altitude change of 1500 km would allow a payload increase

Table 11

Summary of Fields, Particles, and Aeronomy Orbiter Instruments

Instrument	Weight (lbs.)	Power (watts)	Volume ($in.^3$)
Ion-mass spectrometer	3.0	1.0	330
E^1/ion-retarding potential analyzer	2.5	3.0	100
Electron temperature probe	2.0	1.5	90
Neutral-mass spectrometer	10.0	9.0	500
Neutral pressure gauge	7.0	3.0	300
Solar-wind plasma probe*	7.0	5.0	200
Magnetometer*	5.0	1.0	130
Plasma waves (ac electric field) sensor*	2.5	1.5	100
Total	39.0	25.0	1750
Available on bus (1976-77 mission)	50.0	25.0†	2750

*Also used for measuring during Earth-Venus transit.
†When the bus is in sunlight 50 watts is available.

of roughly 35 pounds for a total of 85 pounds for the 1976-77 mission described here. Increasing the apoapsis distance to 50,000 km, see Figure 6, allows an additional 60 pounds to be injected into Venus orbit, but some of this increase must be allotted to additional bus structure and increased battery weight. Lowering of the periapsis to 200 km at injection will also effect a payload increase. It appears that mission-series optimization will involve additional considerations of science objectives, science instruments, orbits, orbit maneuvers, and a science instrument capacity of from 50 to possibly 130 pounds.

4.4 MISSION STATUS

Vast Earth orbiter experience exists both at GSFC and elsewhere in the design and use of this kind of experiment. Those problems specific to a Venus orbiter have been examined in the GSFC Planetary Explorer Phase A report and additional bus design refinement is proceeding under contract (Reference 3). Present technology has been found to be adequate and it is apparent that a number of technology development programs presently in

progress will further enhance the mission. The state of instrument development is such that for each measurement there are several competing flight tested instruments.

5. ATMOSPHERIC CIRCULATION (BALLOON) MISSION

5.1 MISSION SUMMARY

Ten days before encounter two entry vehicles are released from the spinning Planetary Explorer bus, see Figures 17 and 19. The tangential release velocities of the entry vehicles carry them to entry positions 40° and 10° north of the equator and 20° on the dark side of the terminator. The bus impacts the Venus atmosphere and is destroyed. During terminal descent by parachute, see Figure 18, each of the entry vehicles inflates and releases three spherical super-pressure balloons. The six balloons drift with the Venus winds at the 50, 500, and 1200 mb levels and are tracked from Earth to determine the circulation of the Venus atmosphere. The 500 and 1200 mb balloons make measurement of atmospheric temperature and pressure, incoming solar radiation, and outgoing thermal radiation, and transmit this information to Earth.

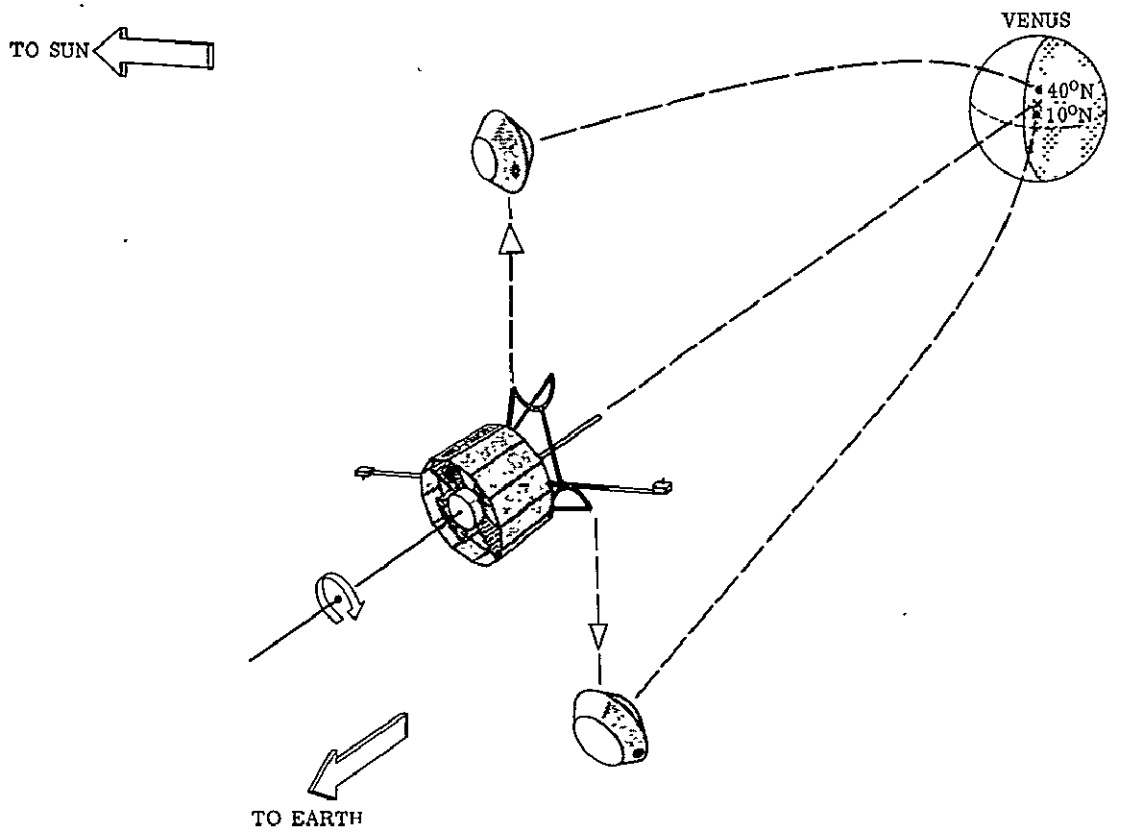


Figure 17. Release two balloon-carrying entry-vehicles at -10 days. Each vehicle contains three balloon-probes.

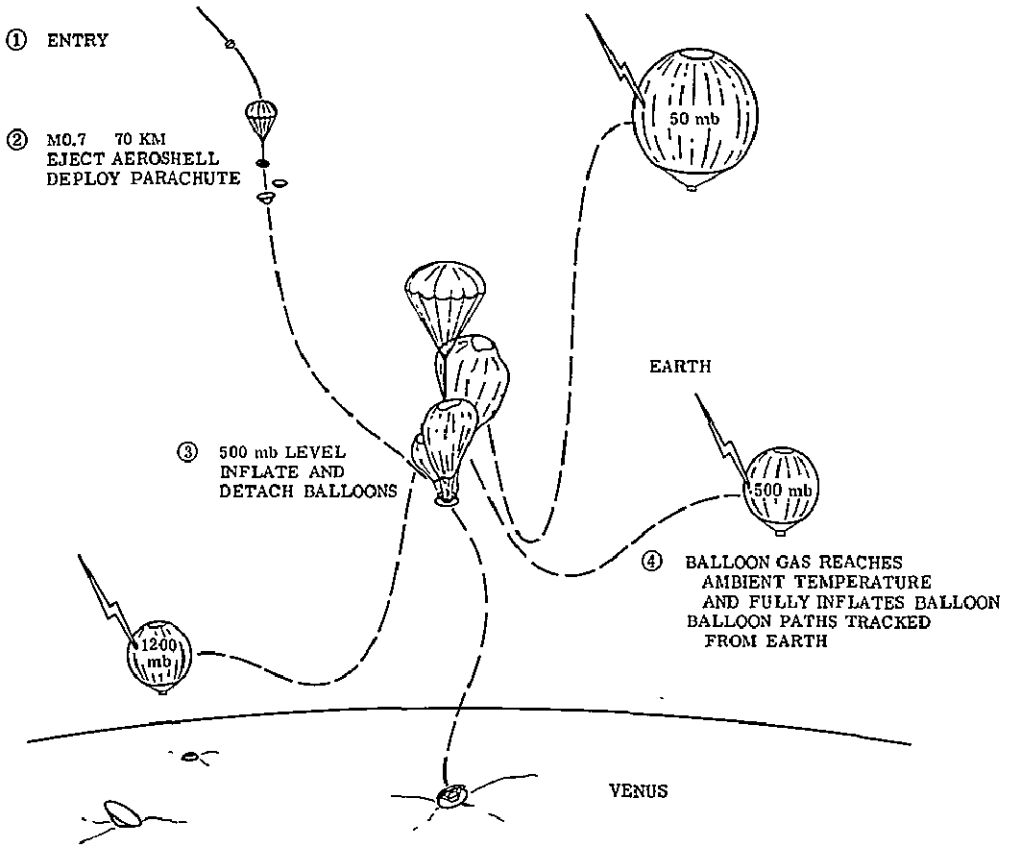


Figure 18. Balloon deployment.

5.2 SYSTEM DESCRIPTION

5.2.1 The Balloons

The inelastic-envelope super-pressure balloon displaces a constant volume of the Venus atmosphere and thus, resisting vertical winds and turbulence, it seeks to float at a constant ambient atmospheric density level. If the balloon drifts across the terminator to the light side of the planet, solar heating causes the gas pressure inside the balloon to increase and unless the balloon skin is of considerable thickness, balloon gas must be vented in order to prevent bursting. As a result, when the balloon again drifts to the dark side of the planet there may be insufficient pressure to maintain its envelope fully extended. If so, the balloon falls and once falling will continue to its destruction in the hot lower atmosphere of Venus. Of the three balloons, the 1200 mb balloon is thermally protected by the cloud cover and its skin is sufficiently thick to allow use of the super pressure required for it to circle the planet. A minor

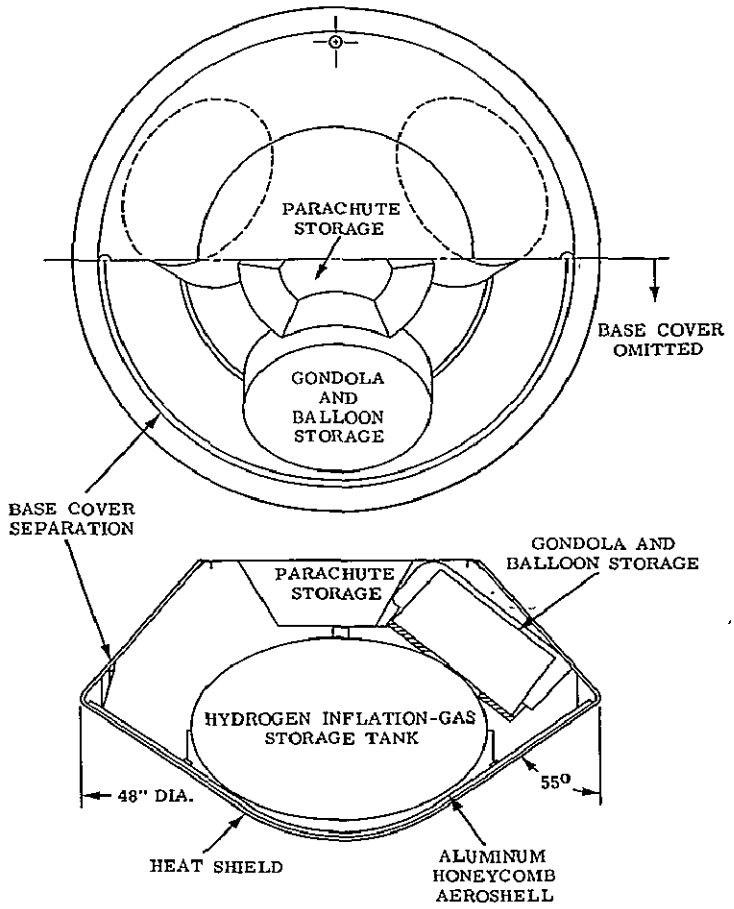


Figure 19. The balloon carrying entry-vehicle configuration. The vehicle carries and inflates three balloons.

increase in balloon skin thickness would also permit the 500 mb balloon to circle the planet, but the 50 mb balloon is exposed to almost full sunlight and a 60% increase in balloon and inflation weight would be required to obtain a planet circling capability.

Upward displacement of the balloons by vertical winds will likewise lead to increased super pressure and require venting. When the vertical wind ceases, the balloon will return to its former level, but if its displacement was large and much venting occurred, there will now be insufficient gas to maintain its envelope fully extended. It will sink to its destruction. Design of the 50, 500, and 1200 mb balloons allows for vertical displacements of 1/2, 1, and 1 km, respectively before excessive venting occurs.

The 500 and 1200 mb balloons are constructed with a Dacron cloth,* to obtain the strength required by superpressure, and a 0.0005" thick Kapton liner to contain the hydrogen inflation gas. Bilaminate construction is utilized for the liner in order to eliminate pin-hole leaks. Due to the low superpressure required for the 50 mb balloon the liner alone is sufficient to supply the required strength. The 13" diameter gondola is attached to the balloon with a Dacron cone.

The hydrogen gas necessary to simultaneously inflate the three balloons is kept at 4500 psi in a single, carbon-filament-composite wound, aluminum lined container. The parachute size is sufficient to limit forces on the balloons to 1 psf during inflation and before separation.

At floating levels above 50 mb balloon system weight becomes prohibitive. Below 1200 mb ambient temperature is too high to allow the use of conventional materials and components.

5.2.2 Communication and Tracking

Communication will occur at an initial distance of $65(10)^6$ km for the 1978 mission opportunity and will be able to be maintained to within 10° of the limb of the planet as viewed from Earth.† The 2300 MHz S band turn-around-transponder on the balloon gondola will radiate 5 watts and will be based on 1975 technology. In order to reduce weight, the gondola with its transponder and science instruments is constructed as a unit and potted in foam rather than employing a conventional modular construction. Because of the low ambient temperature at 50 mb a radio isotope heater is to be used to keep the batteries of the 50 mb balloons at operating temperature. System weights are given in Table 12.

Each hour the receiver on the 50 mb balloon will be turned on and it will switch to one and then to the other of its two antennas, see Figure 20(a), in a search for an interrogation from Earth. Interrogation, however, will normally occur every 5 hours when the receiver, sensing interrogation, will switch to that antenna receiving the largest Earth signal. Coherent transponder operation will begin with a 0.1 minute lock-on time followed for 1.5 minutes by a return of the ranging signal to Earth. The receiver will then switch, if not already connected, to the vertically polarized antenna for a 1.5 minute polarization measurement transmission to Earth. The use of ranging and polarization information for tracking is shown in Figure 20(b). Ranging places the balloon on a circle about the

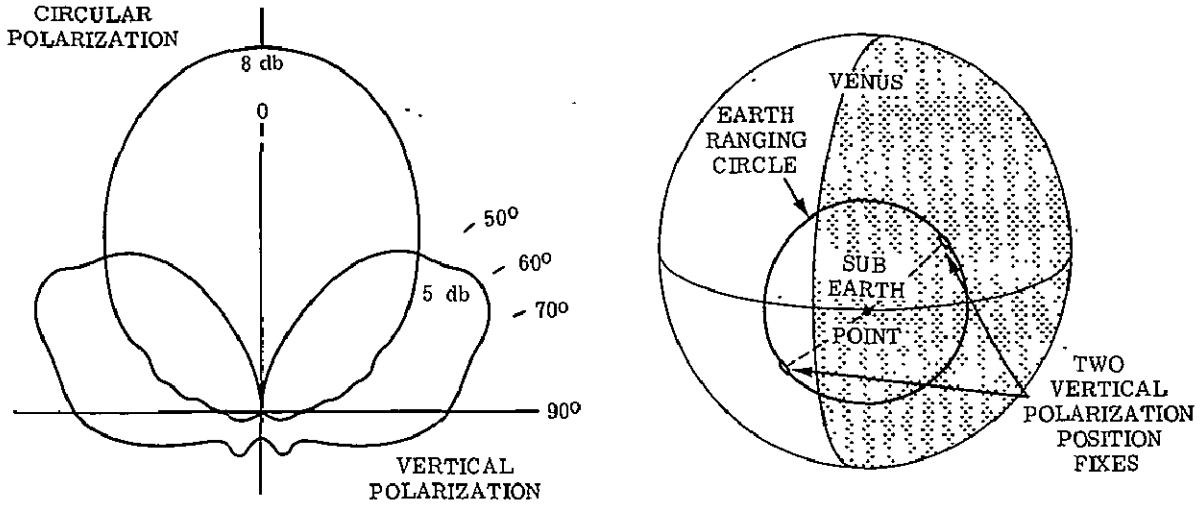
*2.1 and 3.3 oz/sq. yd. respectively.
†Communication and tracking will use the DSN 210 ft. antennas.

Table 12

Summary of Circulation (Balloon) Mission Instruments and Weights for the 1978 Mission Opportunity

Floating level	50 mb	500 mb	1200 mb
Planet radius	6120 km	6107 km	6101 km
Altitude (R_0 = 6050 km)	70	57	51
Instruments	Turn-around-transponder		
		Pressure gauge Temperature gauge Solar flux (incoming) Thermal flux (outgoing)	
		Water vapor	
Balloon diameter	18.5 ft.	10 ft.	7.5 ft.
Balloon and cone weight	5.7 lbs.	6.9 lbs.	5.7 lbs.
Hydrogen gas weight	1.1	1.5	1.2
Gondola weight	16.2	19.0	15.0
Floating weight	23.0	27.4	21.9

Total for balloons for two entry vehicles	2 × 72	144 lbs.
Gas storage tank and accessories	2 × 55	110
Parachute and miscellaneous	2 × 10	20
Aeroshell and accessories	2 × 80	160
Mounting		52
		486
10% contingency		49
Weight mounted on bus		535
Basic bus (includes 10 lb. contingency)		265
Total launched weight		800 lbs.



(a) BALLOON FIELD INTENSITY PATTERN

(b) TRACKING GEOMETRY.

Figure 20. Balloon tracking is accomplished by Earth-based ranging and measurement of the polarization angle of the signal received from the vertically polarized balloon radiation. Ranging error is ± 1.5 km at Earth. Polarization angle measurement error is ± 1°.

sub-Earth point. Polarization measurements identify two possible balloon positions on the ranging circle. Knowledge of the balloon history determines unambiguously which is the true position.

The tracking procedure for the 500 and 1200 mb balloons is the same as for the 50 mb balloon. On these balloons, however, the science measurements indicated in Table 12 are obtained every 15 minutes and stored for replay during the interrogation period which occurs every 5 hours. Data words of 8 bit length are transmitted at 20 bps following the 1.5 minute ranging transmission and before switching to the 1.5 minute vertical polarization transmission. Data transmission requires an additional 1.5 minutes for a total transmission time of 4.5 minutes. Total battery capacity is sufficient for 6 days or 30 transmissions. Depending upon the extent of solar energy penetration as determined from the earlier meteorology probe mission, solar cells may be used to obtain a longer transmission life.

Tracking will permit calculation of the average wind speed between interrogations separated by 5 hours or more. Doppler measurement, however, will permit the measurement to within several centimeters per second of the Earth-directed component of the motion of the balloon over a period of 1 to 4 minutes.

5.3 ALTERNATE MISSION CONFIGURATIONS

The final configuration of the balloon mission is greatly dependent upon the results of the aeronomy and meteorology probe mission. Should this earlier mission determine that emphasis should be placed on examining the temporal and horizontal distribution of the cloud at one or more levels, rather than on atmospheric circulation, the balloon gondola science measurements and float altitude would be modified accordingly. Should the emphasis be on circulation measurements at one or several levels, the gondola science would be removed. A "circulation only" mission might involve entry vehicles targeted to as many as 8 different locations, each carrying an 800 mb balloon with a turn-around transponder and capable of floating completely around the planet.

5.4 MISSION STATUS

The above mission description has resulted from a company sponsored study by Martin Marietta (Reference 6) in cooperation with GSFC. Although the work is preliminary, much of it is based upon an extensive study of buoyant Venus stations (Reference 7).

6. MAPPING ORBITER MISSION

6.1 MISSION SUMMARY

The mapping orbiter, Figure 21, is placed in a polar orbit with a periapsis of 400 km at 45°N. It performs as follows:

a. Radar-altimeter measurement is made of the Venus surface elevation or relief profile along the sub-orbit path from the pole to the equator as shown in Figure 21 and with the resolution shown in Table 13. These measurements, which are made at zero angle of incidence, also provide estimates of the surface dielectric coefficient from which inferences can be made in regard to packing fraction, regolith thickness, and rock classification. Successive altimeter measurement strips are separated at the equator by the rotation of Venus but they converge as the pole is approached, providing dense coverage in this region.

 Radar shadow-mapping of the region enclosed in solid lines on both sides of the sub-orbit path is also obtained by the radar altimeter. A single shadow-map with a known view angle permits rough estimates to be obtained of the elevation of Venus surface features. On subsequent passages, the rotation of the planet surface under the orbit will cause this same region to be shadow-mapped from a different angle. Stereographic techniques will be used with the two mappings to determine the shapes and elevations of surface features. The highly accurate elevation measurements of the altimeter along the sub-orbit path will assist with the stereographic mapping.

b. Passive measurements, using the radar receiver only, are made of the 13.5 cm radiation from the Venus surface to obtain Venus surface-temperature-brightness mapping in the region enclosed within the dashed lines of Figure 21.

c. Bistatic radar measurement will be made of the specular reflection from Venus surface features which are illuminated by an Earth based radar transmitter. The mapped region is shown in Figure 21.

d. Television pictures of cloud-top structure are taken on the light side of Venus from the north-pole-terminator to the south-pole-terminator and from the visible horizon on the East to the visible horizon on the west. Because of difficulty in illustration, only two out of as many as 30 television pictures are shown on Figure 21.

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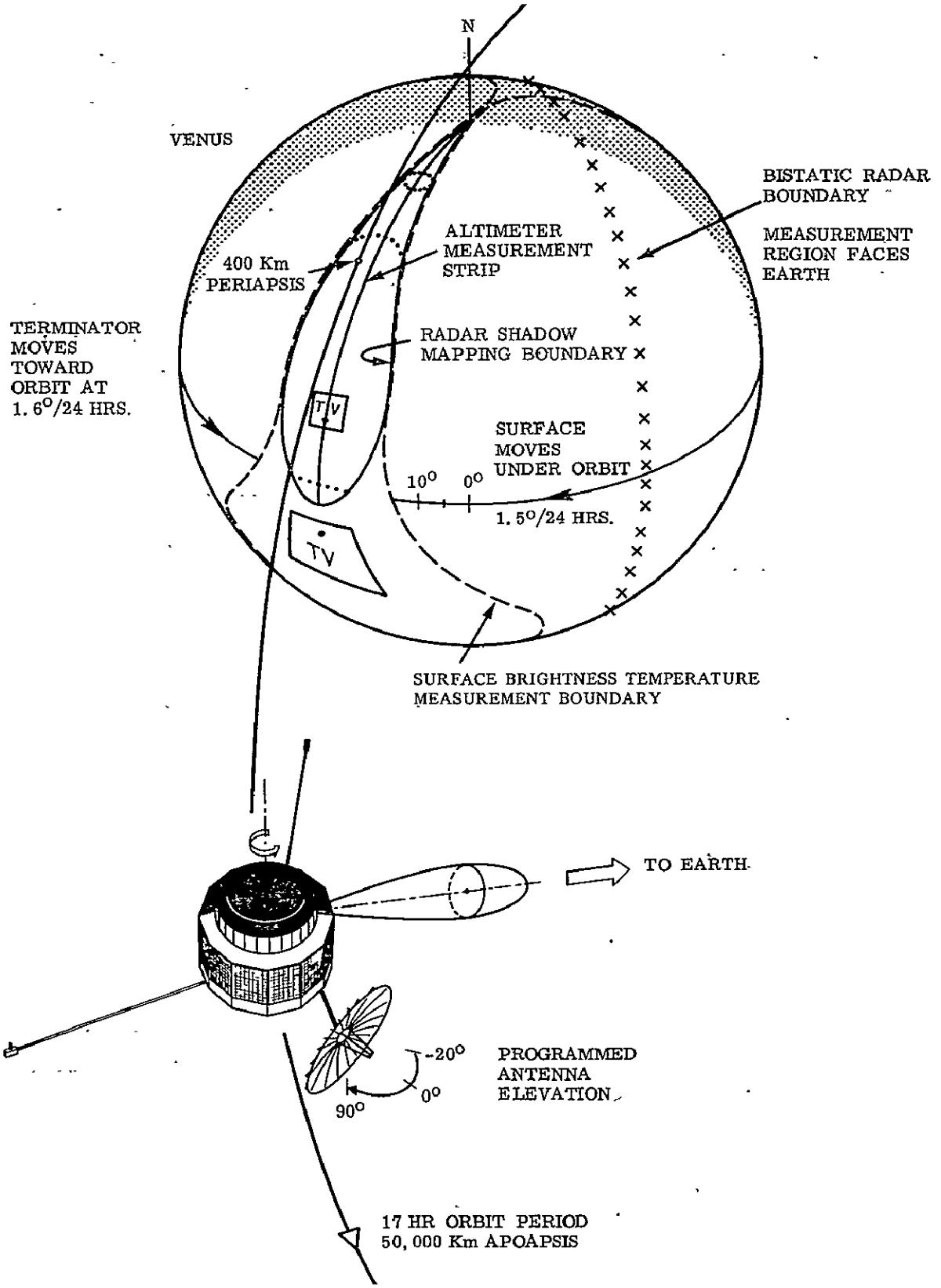


Figure 21. Venus mapping description.

Table 13

Comparison of Mapping-Orbiter-Measurement Resolution and Earth-Based-Measurement Resolution

Measurement	Horizontal Resolution	Remarks
TOPOGRAPHY Orbiter radar-shadow mapping, pole to equator	~1/2 × 100 km	Vertical resolution 0.25 - 0.5 km
Orbiter radar-altimeter	Detects Venus elevation differences of 20 meters based on averaging over 15 km dia. circles at the sub-orbit point from pole to equator.	
Bistatic radar specular reflection mapping	100 - 200 km	Maps on Earth side
Earth based radar 1969 (measured) 1976 (expected)	 ~100 km (35°S to 35°N) ~5 km (10°S to 10°N)	Vertical resolution ~0.3 km ~0.15 km
SURFACE BRIGHTNESS TEMPERATURE Orbiter radar receiver, 13.5 cm wavelength	65 km at 600 km	Temperature resolution 1 - 2K 6° beamwidth
Earth-based radio interferometry at 11.1 cm wavelength 1969 (measured) 1976 (estimated)	 1500 × 5000 km* 1500 × 5000 km	Temperature resolution (model dependent) ~7K ~4K
VISUAL IMAGING Vidicon television on orbiter	2 km at 600 km	45° field of view
Telescope		At quadrature (10^8 km) and based on angular resolution of:
(a) on Earth	>100 km	0.25 arcsec (atmosphere limited)
(b) In Earth orbit	>20 km	0.05 arcsec 72" aperture (diffraction limited)

*Based on 8 sec. of arc along the equator, 24 sec. of arc in a polar direction, and at inferior conjunction.

6.2 ORBIT SELECTION

Orbit selection is based upon obtaining mapping coverage and sufficient payload.

Because Venus is nearly spherical, an upper limit to orbit precession of less than 0.004°/day is obtained and thus precession cannot be used to obtain mapping coverage. We must depend upon the fact that the Venus surface rotates under the orbit with a 243 day siderial period. The relation between orbit periapsis position and radar mapping coverage is shown in Figure 22.

In Figure 23 we see that large orbit payload capacity can be obtained by increasing the orbit apoapsis altitude. A slight additional increase in orbit payload capacity will also result from a decreased periapsis altitude. For the 1976-77 mission considered here the 156 lb. payload requires an apoapsis of 50,000 km. Table 14 presents the weights for this mission.

6.3 MAPPING INSTRUMENTS

6.3.1 Radar

The radar has a range of 1500 km* and furnishes 10 kw 300 ns pulses at a rate of 375 pulses per second. The radar antenna, shown in Figures 21 and 24, is programmed in elevation to look at the sub-orbit point during each bus rotation. Beam path during rotation is shown by the dotted lines in Figure 21.

A temperature calibrator is furnished to enable the radar receiver and the 5 ft. diameter antenna to be used for determining the Venus surface brightness temperature by measuring Venus 13.5 cm radiation.

The radar receiver will also be used on the Earth side of Venus for bistatic operation in conjunction with a separate S-band turnstile antenna system and other electronic adjuncts.

Radar receiver data storage will use a modified Nimbus HDRSS tape recorder with 5 track storage totalling $6(10)^8$ bits and with a maximum input rate of $2(10)^5$ bps. Two tape recorders are included. The second recorder will be held in reserve.

*In order to insure good measurement data, radar range is based on a weak diffuse reflection from the surface. In the event of a high degree of specularity, radar range can be extended to include apoapsis altitudes.

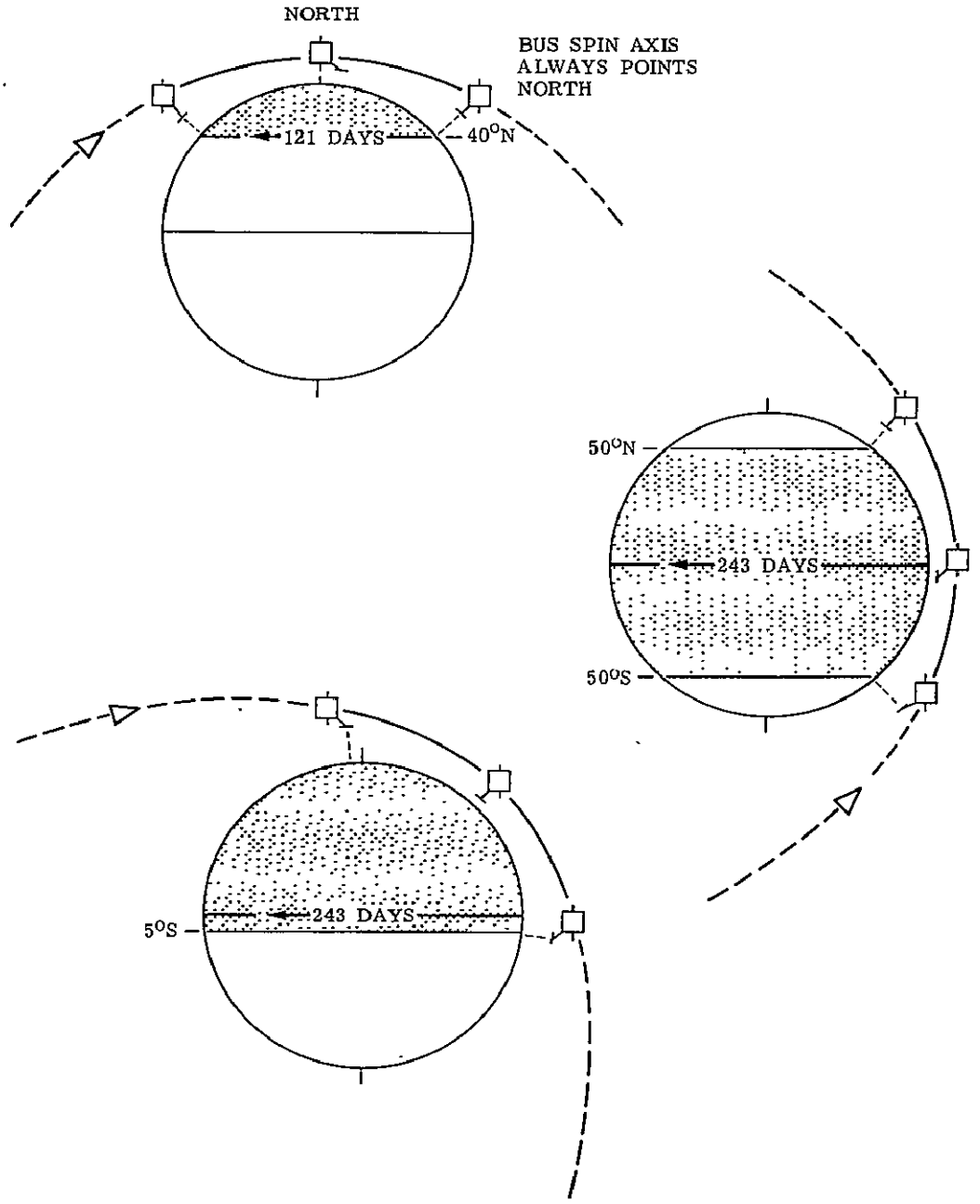


Figure 22. Venus radar-mapping coverage for periapsis placement at 90°N, 0°N, and 45°N. Coverage depends upon the fact that the Venus surface rotates under the orbit with the 243 day sidereal period.

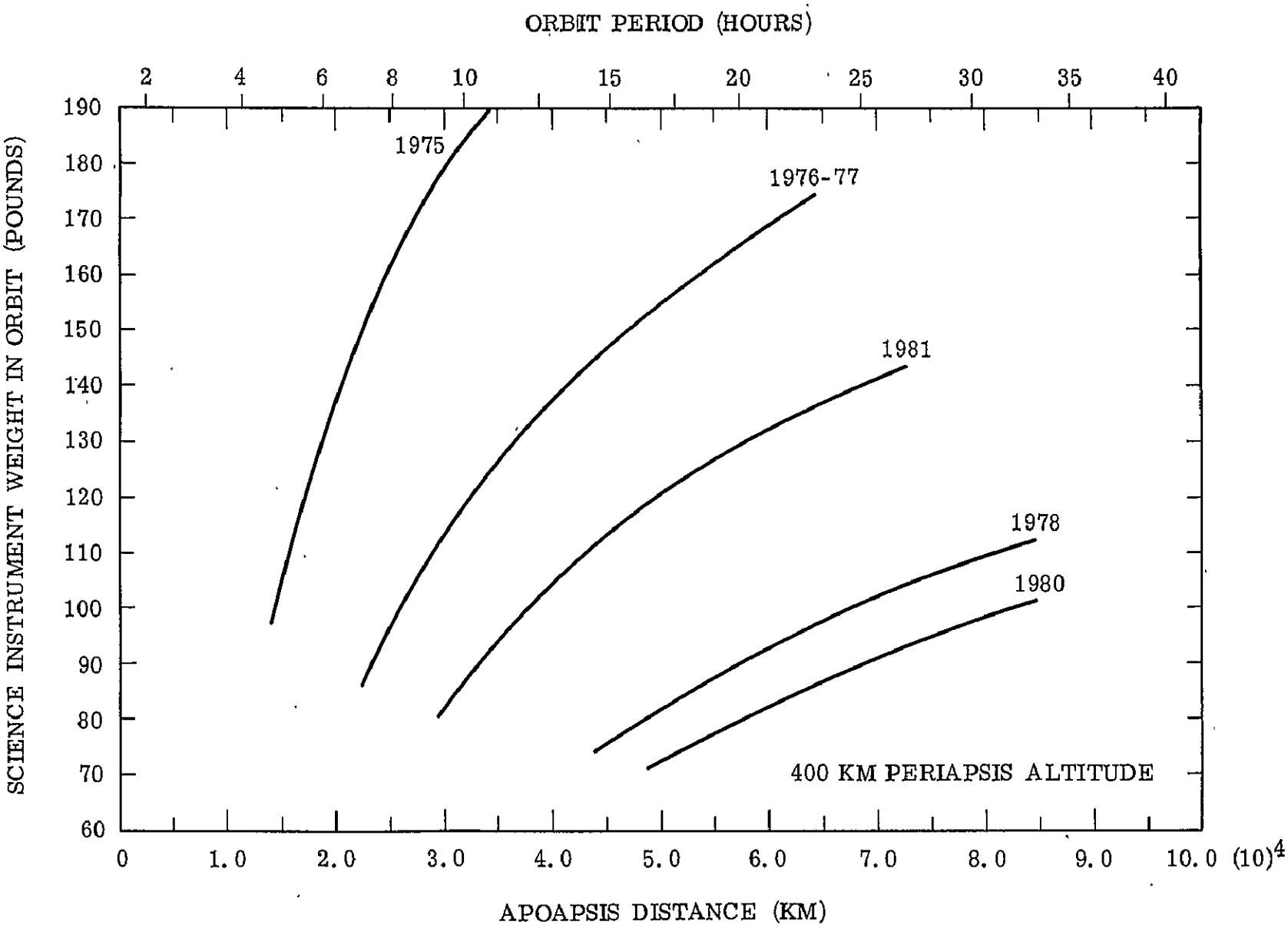


Figure 23. Averaged values for payload in orbit as related to apoapsis altitude for the mapping orbiter. Reduced periapsis results in slight additional payload capacity. For the mapping orbiter no provision is needed for in orbit periapsis change.

Table 14

Summary of Mapping-Orbiter-Mission Instruments and Weights for the 1976-77 Mission Opportunity

Item	Watts	In.^{3}	Lbs.	Total Lbs.
Monostatic radar				
Transmitter	34	300	17.8	
Receiver	6	220	13.2	
A/D converter	2	40	2.0	
Two tape recorders	15*	1180	30.0	
5 ft. dia. antenna, mounting, and elevation drive	2		29.0	
	59†		92.0	92
Bistatic radar (accessories)		100		6
Surface - temperature-measurement calibrator	2	40		2
Vidicon TV	20	850		30
		2730‡		130
20% contingency				26
				156
Planetary Explorer bus (with propulsion)				590
Lift off weight				746

*One operating at any time.
†Maximum battery power required for shadow operation.
‡Available Planetary Explorer module volume 2740 in.^{3}.

6.3.2 Television

The vidicon television has a 1.5 ms. shutter speed, 800 line resolution, and will present an output with 32 shades of grey. Field of view is 45° and at right angles to the bus spin axis. Television data will be stored on the radar tape recorder.

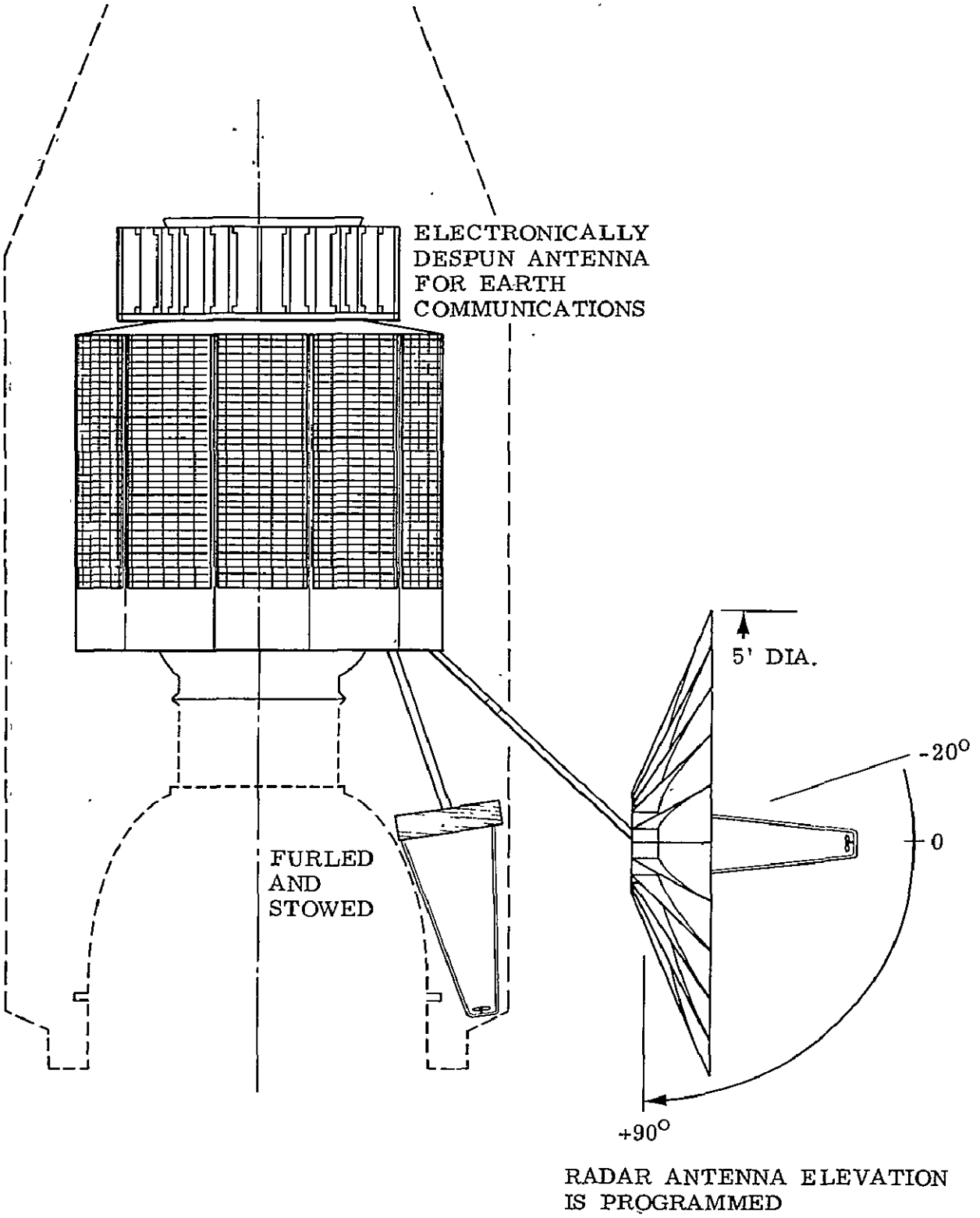


Figure 24. Radar antenna deployment.

6.4 STORAGE, COMMUNICATION, AND COVERAGE

Mapping missions are characterized by the requirement for storage, transmission, and processing of large amounts of information. For our first mapping-mission construction we have selected a storage capacity of $6(10)^8$ bits and use the P.E. electronically despun antenna to communicate to the DSN 210' antennas. Figure 25 summarizes the mapping orbiter storage, communication data, and coverage relations.

a. Mapping days vs. Earth-Venus separation distance is shown.

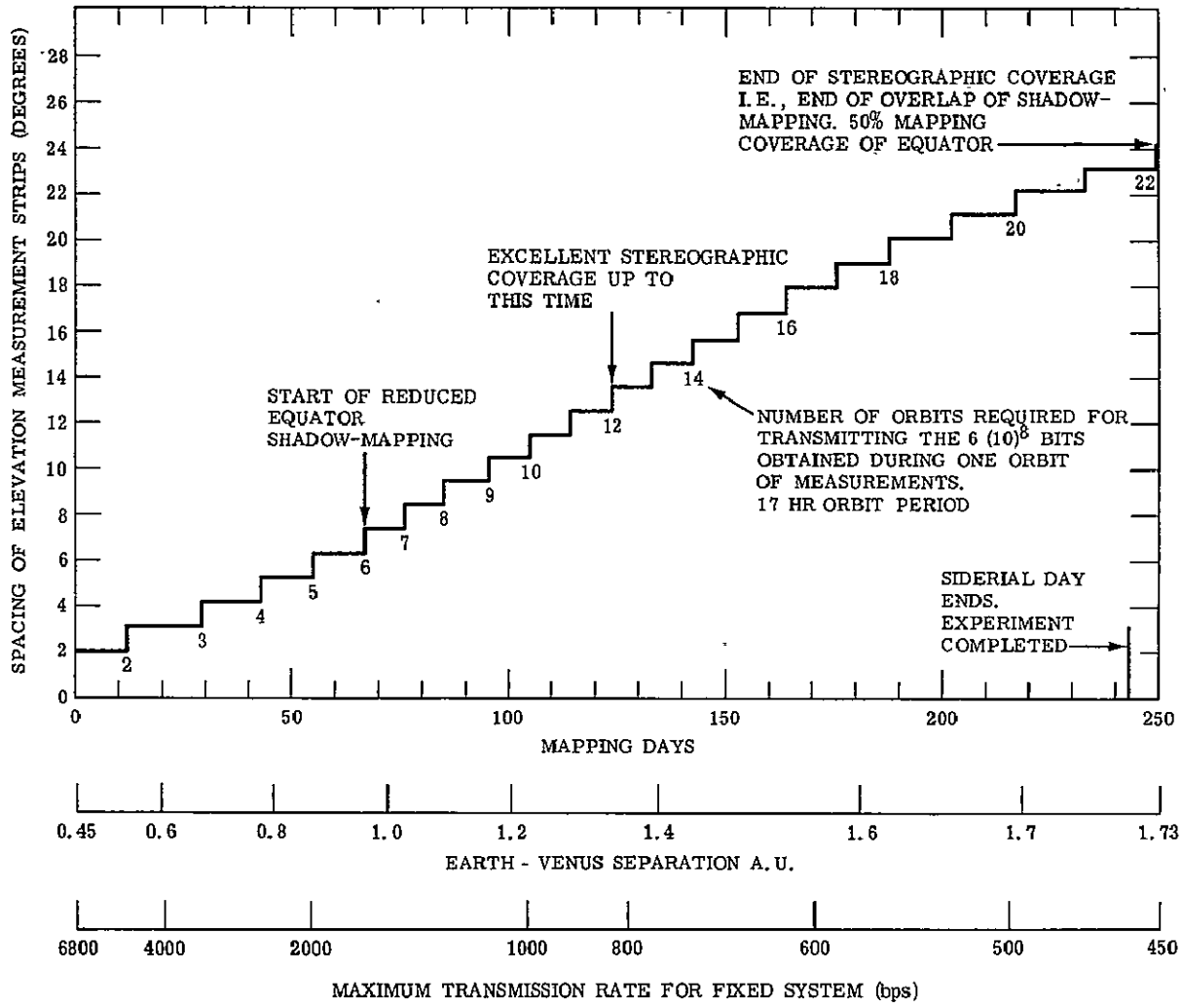


Figure 25. A summary of storage, communication rate, and coverage relations is shown. See text for explanation.

b. Maximum transmission rate for a fixed communication system is shown vs. Earth-Venus separation distance and mapping days.

c. As the maximum transmission rate goes down, the number of hours (or orbits) required to transmit the $6(10)^8$ bits of stored data increases.

d. As the number of "transmission only" orbits increases, the spacing between elevation measurement strips on the Venus surface increases.

It can be seen, for example, that doubling the maximum transmission rate or halving the total data stored for transmission, would halve the separation of the elevation measurement strips and insure stereographic radar mapping throughout the entire mission; or, if coverage were held unchanged, the DSN effort could be reduced from continuous reception to half-time reception.

A more advanced system configuration might replace the present fixed mounted 5' diameter antenna with a mechanically despun antenna with a programmed elevation control. In this configuration the same antenna would be used for both radar observation of Venus and for communication to Earth. While general stereographic coverage would remain unchanged a number of advantages would accrue:

a. Altitude measurement strips would be able to be obtained from 90°N to 30°S, or from 65°N to 65°S.

b. It would be possible to examine select objects with a 2 km by 2 km resolution by using the system as a coherent, side looking, synthetic aperture, unfocussed radar.

c. Communication rate to Earth would increase by a factor of eight. Total monitoring time by the DSIF would be reduced by a factor of eight.

6.5 MISSION STATUS

The above mission description incorporates the results of radar studies (Reference 8) furnished for the mission series by H. J. Goldman* and D. L. Roberts.*

*Astro-Sciences, IIT Research Institute.

While only rudimentary study has been made of the surface temperature, television, and bi-static radar measurements and of the structural, thermal, system operation, and orbit problems specific to a mapping mission, no unusual or disabling problems have been identified. Considerable related study and experience exists for each of the proposed measurements and for the required systems.

7. VENUS INTERIOR EXAMINATION (SEISMIC LANDER) MISSION

7.1 MISSION SUMMARY

Thirty minutes before encounter three entry probes are released from the spinning bus. The two outer probes are released tangentially and the center probe is given a small forward separation speed. The three probes enter the atmosphere near the sub-Earth point and descend together as shown in Figure 26. The bus is destroyed upon entry. The bomb probe reaches the surface serveral minutes before the two seismic probes and awaits their landing. The two seismic probes deploy parachutes at 0.5 km above the surface; they land gently, one and two km respectively from the bomb, and uncage their seismometers. The geometry of the landed positions is shown in Figure 27. Three minutes after the seismic probes land, the bomb explodes, starting the active seismic experiment. In the first 20 minutes after the explosion the seismic landers transmit the results of the explosion measurements to Earth while at the same time storing for replay the measurements obtained from any natural Venus quakes which occur during this 20 minute

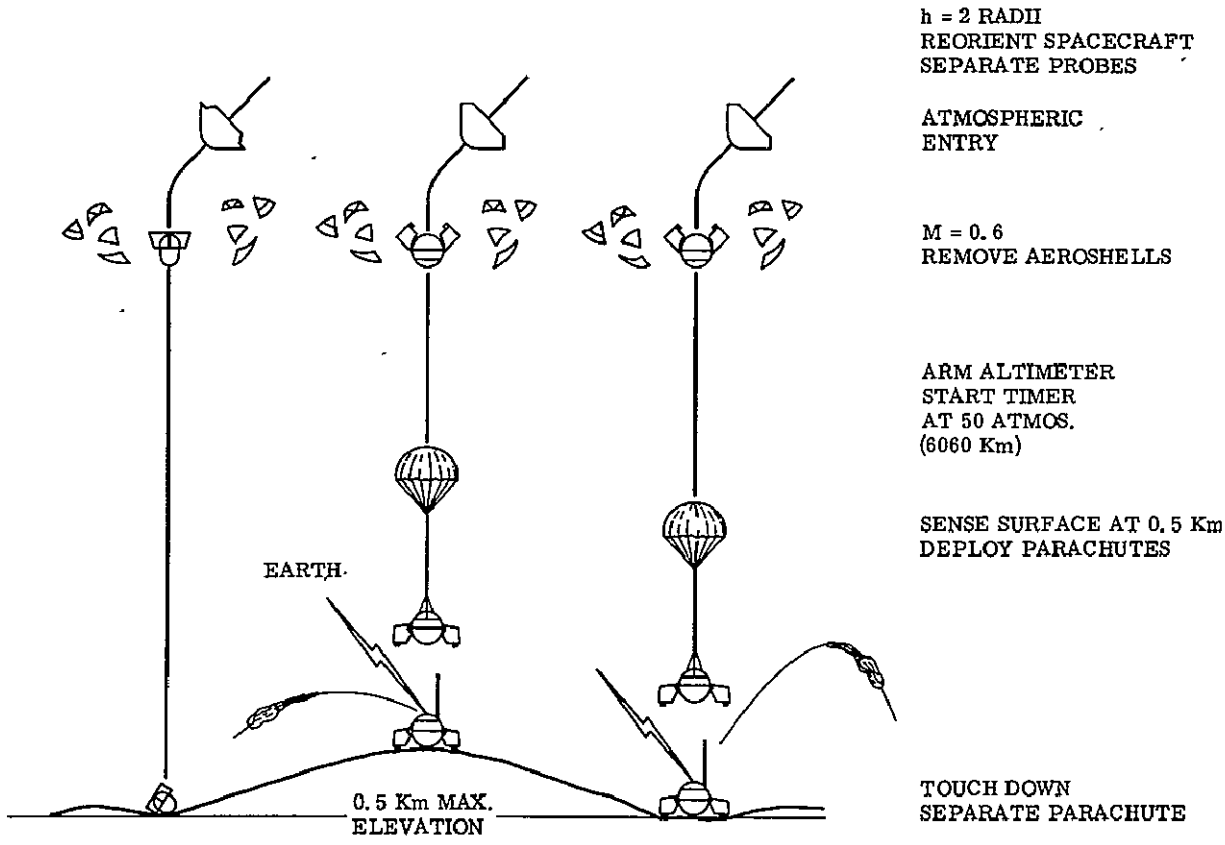


Figure 26. Bomb and seismometer deployment.

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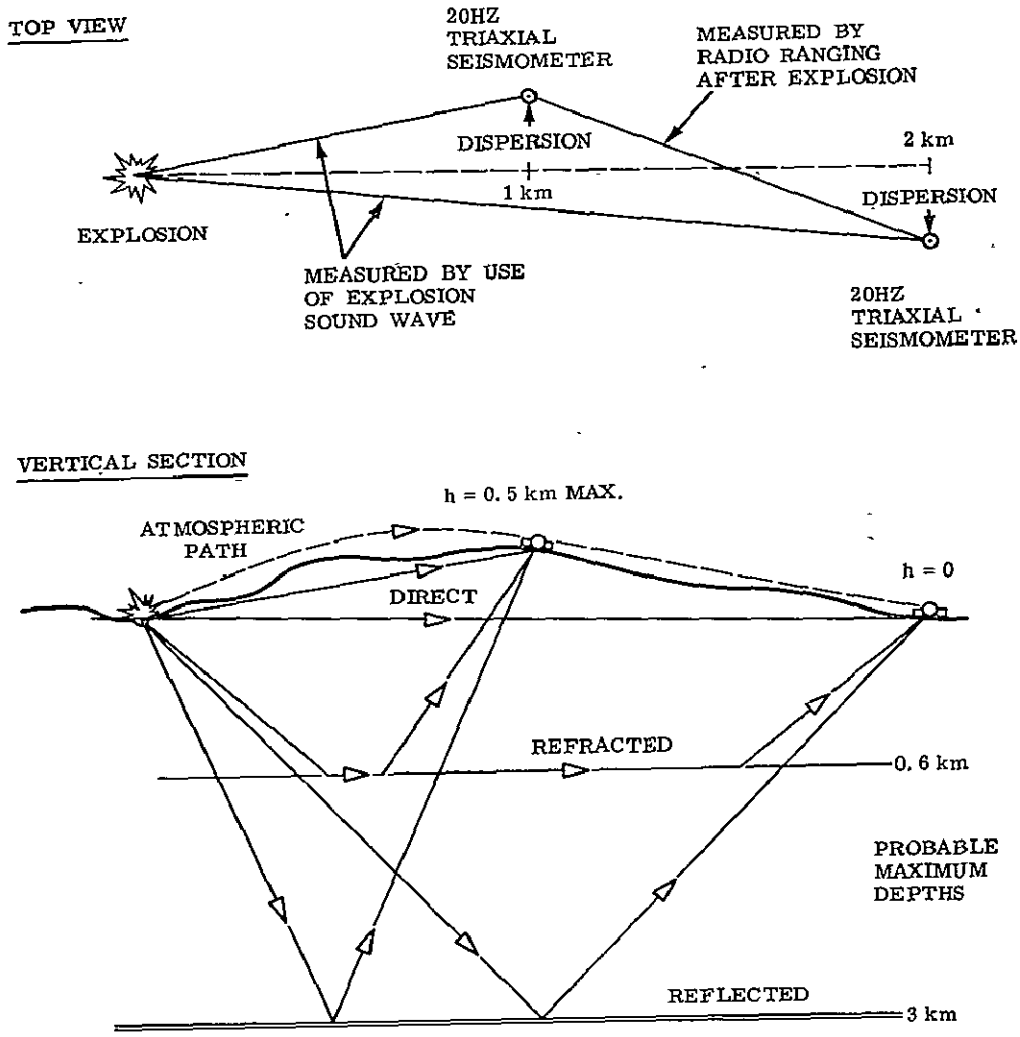


Figure 27. Geometry of the Venus active seismic experiment.

transmission period. During the final 40 minutes of lander life, Venus quakes and atmospheric noise are monitored in real time.

The active seismic experiment is intended to determine the surface structure, compressional and shear wave velocities, and the elastic and anelastic properties of the surface material down to a depth of about 0.6 km. The seismometers are also used during the probe life-time, to measure nearby micro-earthquakes and natural seismicity. It is possible that significant Venus quakes will occur and provide information about the planet's interior structure. Average and peak sound levels are also monitored.

7.2 TARGETING

The placing of the bomb and seismometer probes on the Venus surface is designed to obtain a minimum dispersion about the relative positions selected for the landed probes. The probes are separated from the bus simultaneously with identical spin rates; they enter the Venus atmosphere at the same time with large and equal entry angles and have the same hypersonic ballistic coefficients. Studies have shown that for these conditions, relative dispersion during entry can be expected to be less than 20 meters.

The subsonic ballistic coefficients of the probes are equal and large. The probes will pass rapidly through the high atmosphere where large and variable winds may exist which, if allowed to act over long periods, might cause large relative dispersion.

Because of equal descent speeds, however, all probes will always be at the same atmospheric level and thus dispersion will depend only upon the difference in wind speed over a horizontal distance of at most 2 km rather than upon the wind speed itself. Since the winds near the surface are small, changes in the horizontal distance between the landed bomb and the seismic probes due to the 2-3 minute parachute descent of seismic probes time is not expected to be significant.

Total relative dispersion is not expected to impare the active seismic experiment.

Knowledge of the landed geometry of the seismic experiment will be obtained by use of:

a. pressure measurements to determine local and relative altitudes;

b. microbarograph sound-arrival-time detection, and pressure and temperature measurement, to determine the over-the-surface distance of the seismometers from the bomb; and

c. a turn-around, 24 MHz, 5 watt radio-ranging device on the seismometer probes to determine their over-the-surface separation.

The sides of the landed position triangle will be measured to better than 1% if high surface features do not separate the landed probes.

7.3 THE BOMB

Because of present uncertainty in the ability to penetrate the Venus surface to a sufficient depth to achieve good explosion coupling, the bomb, Figure 28

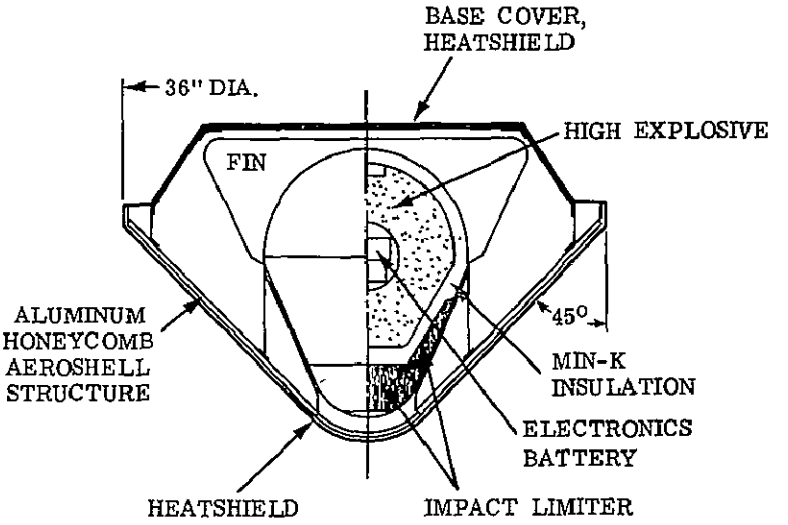


Figure 28. Seismometer experiment "bomb" probe.

and Table 15, is designed for explosion on the surface. The explosion of the bomb in proper relation to seismometer readiness is determined by timers in both the bomb and the two seismometer probes. These timers, which are started simultaneously at the time of release of the probes from the bus vehicle, make allowance for surface elevation differences and are used for exploding the bomb and for programming the seismometer operation as shown in Table 16.

Thermal control of the bomb is less a problem than for the landers because its temperature limit is much higher, it has a much larger heat capacity, and its operating lifetime is much shorter. The bomb is designed so that the detonator and timer electronics are packaged in a small pressure vessel placed in the center of the explosive in order to take advantage of the heat capacity and insulating properties of the explosive. Thermal control is achieved by placing Min-K 1301 insulation on the outer surface of the bomb casing to prevent the TACOT surface temperature from rising above 600°F prior to detonation.

7.4 SEISMOMETER LANDER

It can be seen from Figures 26, 29, and 30 that before parachute deployment the seismic probes are upside down with the communication antenna looking downward. During this portion of the descent the communication transmitter is pulsed at 5 watts peak power and used with a rudimentary homodyne receiver to detect the approaching surface. At 0.5 km above the surface an 8 ft.

Table 15

Bomb Weight Summary

Explosive (TACOT)		50.0 lbs.
Electronics and timer		2.1
Insulation		2.8
Electronics installation (pressure vessel and support)		1.9
Explosive container		7.0
Pressure equalizer (bellows and adapter)		.8
Insulation cover		3.8
Aerodynamic fins and installation		1.6
Shock attenuator and installation		.9
Impacting weight:		70.9
Entry aeroshell		40.5
Structure	22.5	
Heat shield		
Front cone	13.0	
Base cover	5.0	
Yo-yo despin system		1.3
Pyrotechnic system for jettisoning aeroshell (including structural adaption)		6.0
Entry weight:		118.7 lbs.

Table 16

Mission Sequence of Events for the Assumption of a Level Landing Region at a Radius of 6050 km

Time (minutes)	Event	Notes
0	Start timers on 3 probes	Spacecraft command
2 sec	Separate probes from bus	Spacecraft command
24	Despin probes before entry	B.C. = 0.475 slugs/ft.^{2}
30	At M 0.6 sense 1 g decreasing, separate aeroshell	B.C. = 3.8 slugs/ft.^{2}
66	Activate altimeters on the seismic probes	P = 50 atmospheres R = 6060 km
76	Deploy chute on seismic probes	B.C. = 0.22 slugs/ft.^{2}
77.5	Bomb impact	V = 13 m/s
79	Seismometer touchdown Deflect chutes	V = 3 m/s
80	Begin seismometer uncaging	
81.5	Transmit signal with pressure, temperature, and engineering data	30 seconds allowed for Earth station lockon.
82	Bomb detonation. Start active data storage, 3. Continue transmission of P, T, E.	Timers in probes and bomb.
82.25	Microbarograph active data complete	
82.5	Seismometer active data storage complete. Initiate active seismic data transmission. Deploy whip antennas for range measurements, make range measurements and store them.	30 bps
83.5	Range measurements complete. Begin passive seismic data storage.	
104	Active seismic data transmission complete. Readout range data and transmit.	Requires 130 sec
106	Range transmission complete. End of active experiment. Readout stored passive data.	Requires 3 sec

Table 16 (Continued)

Time (minutes)	Event	Notes
106	Passive data transmission complete. Begin real time monitoring of: seismometers microbarographs pressure temperature engineering data	 30 bps
140	Design mission complete. 130°F	Surface operation of one hour at 6050 km

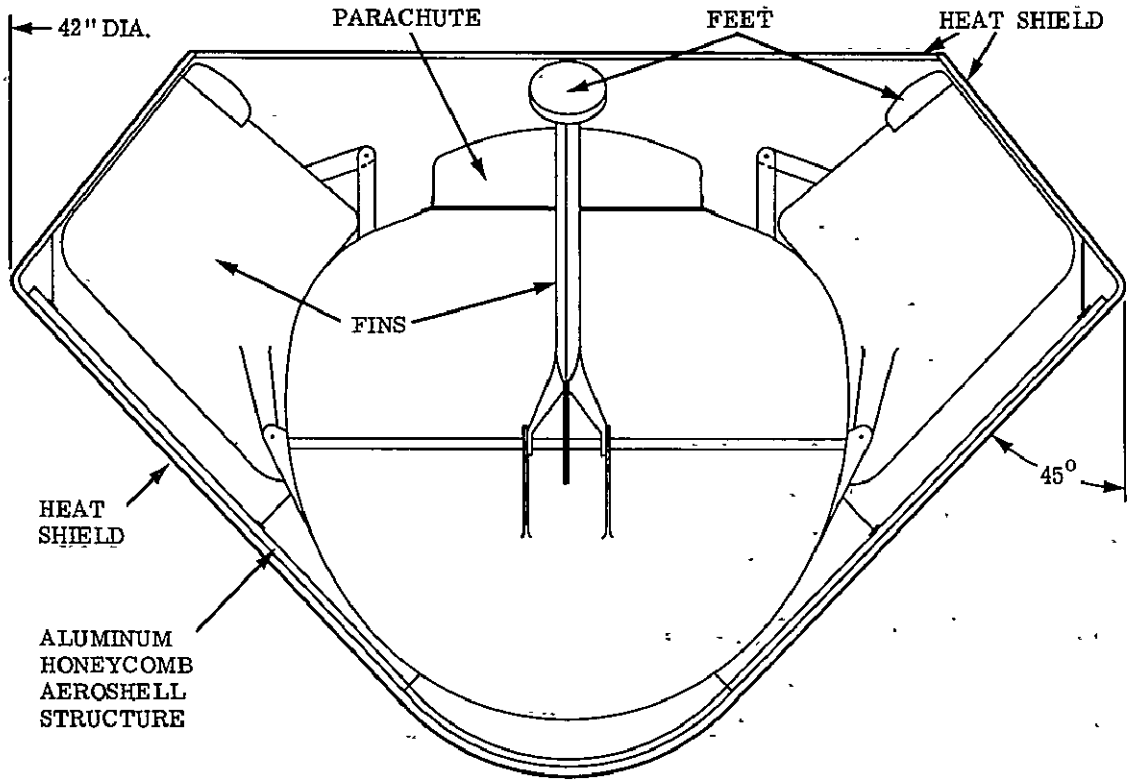


Figure 29. Seismometer probe entry and descent configurations.

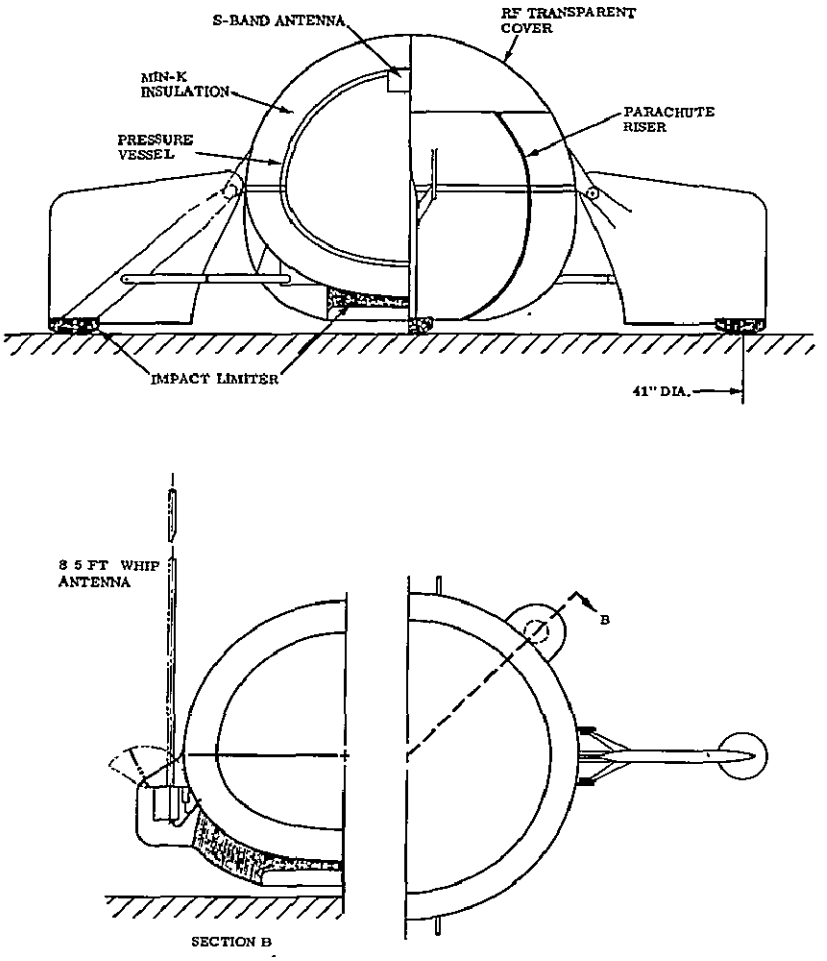


Figure 30. Seismometer probe landed configuration.

diameter glass-fabric parachute is deployed, the seismic probe is up-righted and lowered to the surface at low speed. At touch-down the parachute is released and deflected by means of a small rocket, see Figure 26. Seismic lander construction is described in Figure 30 and Table 17. The whip antenna used for radio ranging between the two seismometers is shown in Figure 30.

The characteristics of the three axis seismometers are primarily based upon the descriptions of seismic instruments for a Viking Mars lander. A three axis unit is used since lunar experience has shown that much information would be lost should only a single axis unit be used. Since for short path distances there is little damping and high frequencies prevail, a 20 Hz response is used. Seismometer design will allow operation in the presence of lander tilting of 25° or at higher angles with reduced sensitivity.

Table 17

Seismic Lander Weight Summary

Internal equipment		28.9 lbs.
Electronics and battery	21.3	
Science	3.1	
Seismic instrument	4.5	
Pressure shell		17.0
Waffle shell	14.2	
Closure flanges and hardware	2.8	
Internal mounting structure and penetrations		5.8
Insulation (Min-K) and heat sink material		9.8
Insulation cover and closure joints		5.0
Landing legs, fins, and mechanisms		10.0
Shock attenuator and installation		2.5
Parachute system		5.0
Chute	2.5	
Metallic harness and riser, fairing, pyrotechnics and rocket	1.5	
Parachute can, cover, its deployment and landing legs lock/release	1.0	
Landed weight:		84.0
Entry aeroshell		59.0
Structure	32.0	
Heat shield		
Forward cone	20.0	
Base cone	7.0	
Despin Yo-Yo system		1.5
Pyrotechnic system for jettisoning aeroshell (including structural adaptation)		7.5
Entry weight:		152.0 lbs.

Table 18

System Weight Summary

Seismic landers 2 × 152	304
Bomb	119
	423
Contingency (10%)	42
	465
Adapter structure and deployment mechanism	75
Contingency (10%)	8
Spacecraft	266
Total launched weight	814

7.5 COMMUNICATION

The seismic landers both communicate directly to Earth at 30 bps by means of a 5w S-band, non-coherent system using a 5 db crossed-slot antenna.* Data encoding is at 8 bits per word. The first 30 seconds of seismometer data following detonation is sampled at a high rate and stored for an immediate 20 minute readout at the 30 bps transmission rate. During this 20 minute readout period for the active seismic experiment data, passive seismic data is stored and processed for transmission in compressed form at the end of the 20 minute period. The last half of the 1 hour mission is used to transmit real time data taken at the low rate of two samples per second.

7.6 DISCUSSION

The final configuration of the seismometer experiment may depend heavily upon our knowledge of Venus obtained from preceding missions:

a. The meteorology probe mission will determine the severity of surface conditions by measuring surface pressure and temperature, and obtaining a rough estimate of the magnitude of surface winds;

*Communication is to the DSN 210 ft. antennas.

b. A penetrometer experiment on the meteorology probe or TV lander mission would explore the feasibility of use of a penetrating bomb to obtain increased horizontal and vertical range for the active seismic experiment and to perform a "Terradynamic"* experiment;

c. Measurements of the surface dielectric coefficient during a mapping-orbiter mission may give information as to surface packing fraction, regolith thickness, and rock classification.

If natural Venus quakes are found to be frequent, a follow-on experiment might use a passive array of 2 or perhaps 3 seismic landers constructed with conventional thermal design to last 25-50 hours. If they are infrequent then attention must be devoted to devising the high temperature equipment necessary for lifetimes of several years or more.

7.7 MISSION STATUS

This mission was developed as a company sponsored effort by Martin Marietta (Reference 6). It is the result of a short period, but intensive application of present scientific and technical knowledge to the total problem of Venus interior examination.

*The estimation of surface properties down to the depth of penetration by means of examining the acceleration profile obtained during penetration. Transmission of the acceleration measurement is from a trailing wire. This method was developed by the Sandia Corp. and has had extensive Earth use.

8. VENUS SURFACE-INSPECTION (TELEVISION LANDER) MISSION

8.1 MISSION SUMMARY

Following entry into the Venus atmosphere at the sub-Earth position, 7° on the light side of the morning terminator and 4° south of the equator, the lander discards its heat shield and descends to the Venus surface. The lander radar senses the approach to the surface at 1/2 km altitude and causes the parachute and the lander legs to be deployed, see Figure 31. On touch-down the parachute is released and is removed by means of a

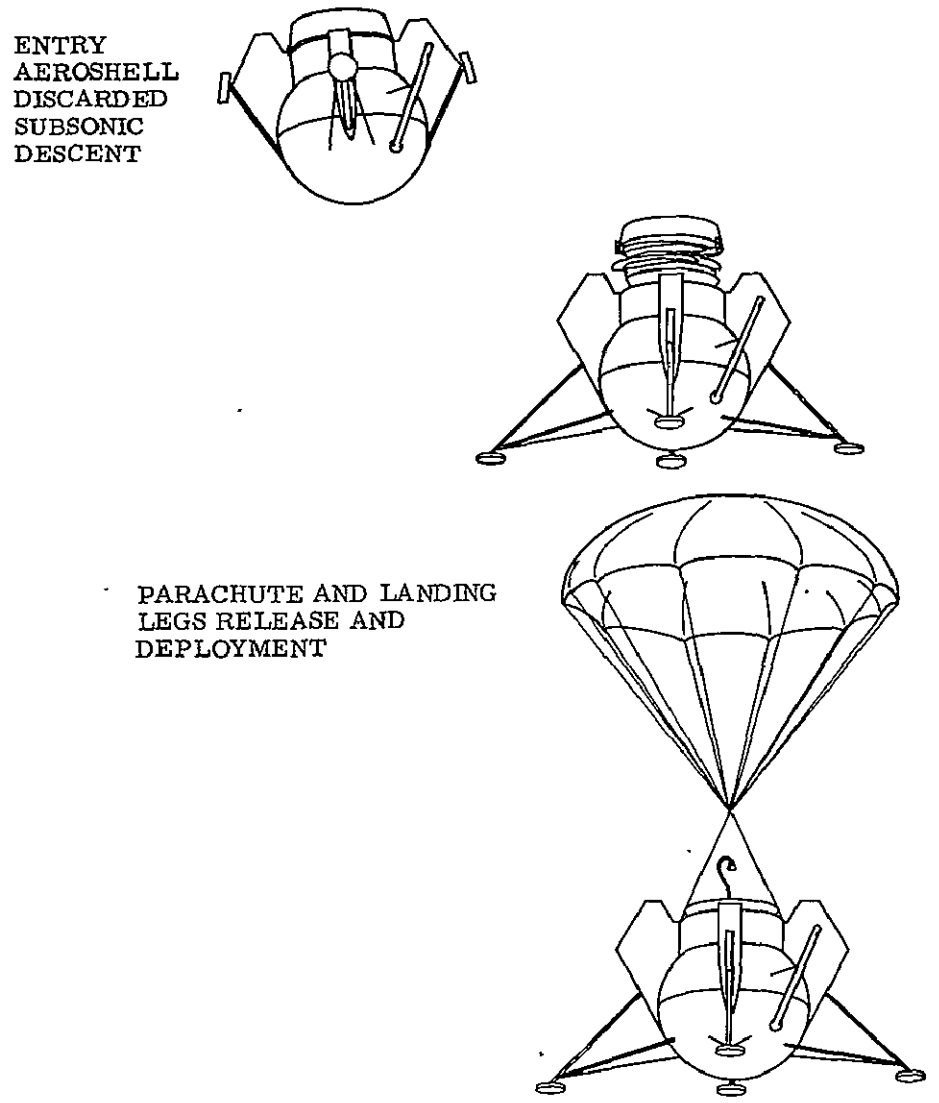


Figure 31. Parachute and leg deployment.

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small rocket. The zenith-seeking high-gain communication antenna, Figure 32, is uncaged and the soil sampling drill and the TV periscope, Figure 33, are deployed. The lander, following the schedule of Table 19:

a. chemically analyzes two samples of the surface soil;

b. obtains two microscope images of the soil samples to determine their petrographic nature;

c. takes 10 TV pictures of the surrounding area, see Figure 34, with resolutions of from several millimeters to several tens of meters, and ranges of from one meter to as much as 15 km depending upon lander elevation, local lighting, and atmospheric conditions;

d. performs measurement of ambient pressure, temperature, and wind velocity.

Lifetime on the surface is two hours.

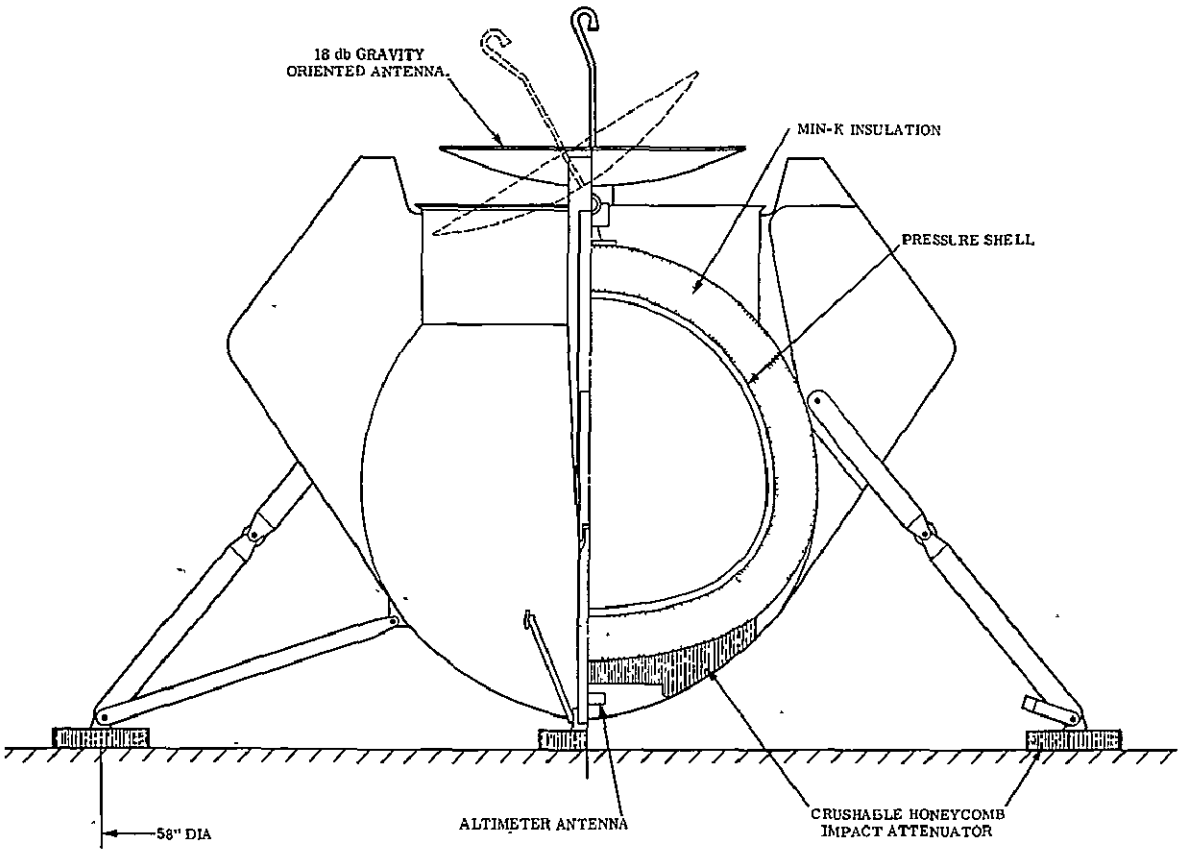


Figure 32. TV lander configuration.

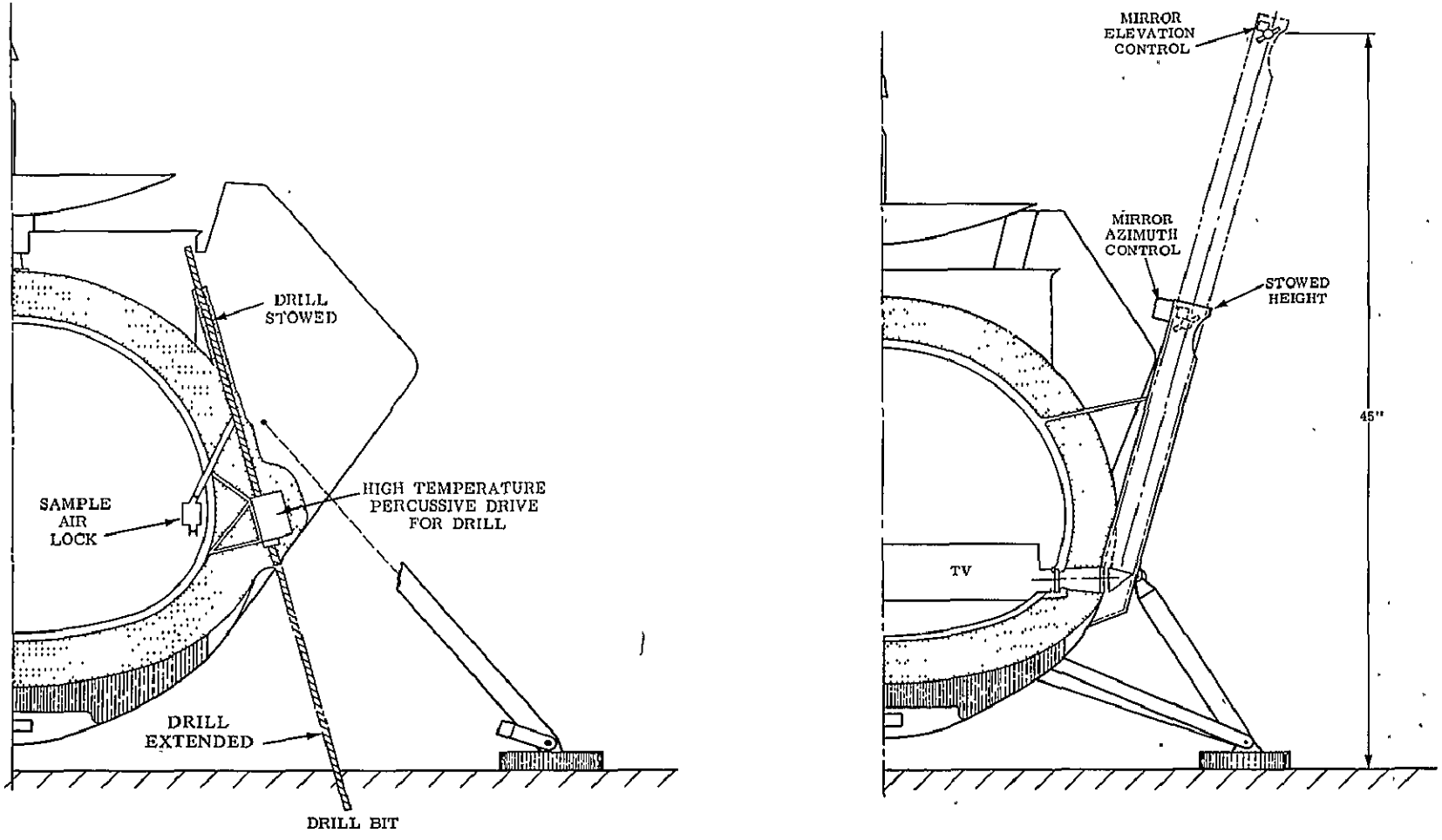


Figure 33. Soil sampling and TV picture mechanization.

Table 19

Mission Sequence of Events

Time (minutes)	Event	Notes
-13	Activate altimeter (timed function)	P = 50 bars R = 6060 km
	Sense surface at 0.5 km Deploy chute Deploy and lock legs	$B.C._{chute}$ = 0.22 sl/ft^2
0	Probe touchdown Release and deflect chute Deploy antenna Deploy TV optics Activate soil sampler	Nominal 6050 km surface radius
2	Take first TV picture	30 seconds allotted
2.5	Begin to collect and transport soil sample Begin first TV transmission	8 minutes required
10.5	End TV transmission Begin transmission of P, T, W, E Take second TV picture Begin soil analysis and store data	
11	Begin second TV transmission	
19	End second TV transmission Begin second P, T, W, E transmission Take third TV picture	Soil sample
20	Begin third transmission P, T, W, followed by composition data from storage	Earth based evaluation of first two pictures possible.
28	Take fourth TV picture Repeat sequence from 2.5 minutes	Second soil analysis complete at 58 minutes.
58	Continue pictures 8 through 12	
100	Complete automatic sequence	

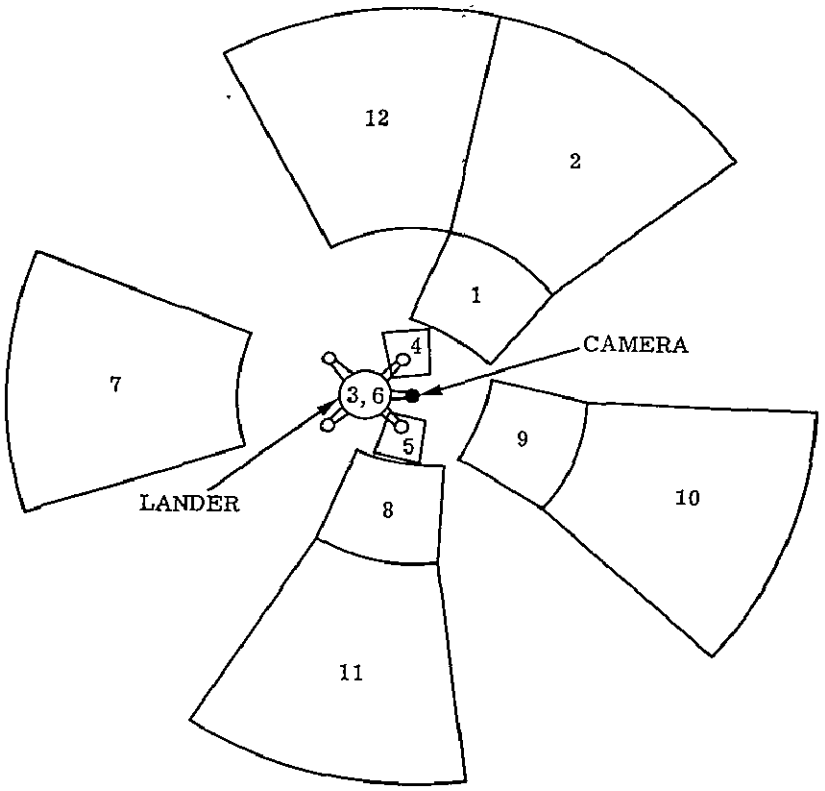


Figure 34. Possible TV picture sequence. Numbers 3 and 6 represents microscope pictures of the soil sample viewed within the lander.

8.2 COMMUNICATION

System design is determined by the requirement for transmission directly to Earth, for command capability, and for high data transmission rate:

a. A sub-Earth landing point is required to minimize atmospheric attenuation and to simplify pointing the lander antenna;

b. The 18 db parabolic antenna is capable of 30° tilt. It uses motor driven gimbals controlled by a sensor within the pressure vessel as a means to point the antenna to the zenith regardless of the tilt of the landed vehicle;

c. A 20 watt S-band transmitter is used to obtain a data transmission rate of 2000 bps to the DSN 210 ft. antennas.

d. A command system is used to permit reprogramming of the picture sequence following examination of the first few pictures.

Since single picture transmission time is 8 minutes and round-trip communication time is 12 minutes, roughly 20 minutes will be required before the automatic control of the lander picture taking sequence can be interrupted by command from Earth. Thus the first opportunity to effect picture control would occur at the beginning of the 4th of the total of 12 pictures.

Three kinds of data are transmitted to Earth: image data, stored mass-spectrometer data, and real time science and engineering data. Image data is read directly from the TV storage tube in analog form for conversion to 6 bit words and placed in format for direct transmission to Earth. Stored spectrometer data (3600 bits) is read out at the end of each sampling period and interleaved with the real time transmission of atmospheric pressure, temperature, and winds and engineering data. Thirty seconds is allowed for this data prior to each picture transmission.

Since pictures on Venus are expected to lack high contrast, the use of TV data compression of possibly 2 to 1 must be considered as a means of doubling the number of pictures returned. The use of data compression would, however, require a real-time reconstruction capability at the DSN in order to provide real-time picture viewing for command purposes.

The radar altimeter required to provide a signal for opening the parachute at an altitude of 0.5 km will be provided by operating the communication transmitter in a pulsed mode and switching its output to a downward looking antenna.

8.3 LANDER INSTRUMENTS

A summary of the instruments is given in Table 20.

8.3.1 Television

The image tube is sufficiently sensitive to operate under moonlight conditions and has a silicon-target imaging tube which can store the image for more than a half hour for read-out at the desired bit rate. Camera exposure will be automatically adjusted by a photometer. Image resolution is 400 lines and each picture element will resolve 64 light levels (6 bits/picture element) resulting in $9.7(10)^5$ bits per picture. Because the camera target must remain cool, light from the area of interest is collected by a mirror mounted in a boom which is extended to 45 inches above the surface. This

Table 20

Science Payload for 1981 TV Mission

	Weight (pounds)	Power (watts)	Dimensions (inches)	Bits/ Sample	Samples/ Mission	Remarks
Panoramic TV						
Camera system	12	14	4 × 6.5 × 12	972 000	10	Operates continuously
Mast and associated optics	5	–	2 dia. × 37	–	–	
Optics	1.5	–	–	–	–	
Motor drives	2	3	–	–	–	Total operating time 15 mins.
Microscope TV						
Microscope, petrographic	6	3	8 × 8 × 3	972 000	2	8 minutes operating time/picture
Connecting optics	1	–	0.5 dia.	–	–	
Sample collection and transport	6	50	small	–	–	
Mass spectrometer surface composition analyzer	20	50	500 $in.^3$	1600	2	Data based on mass-spectrometer analyzer
Atmospheric pressure sensor	0.25	0.2	2 dia. × 1	8	12	
Atmospheric temperature sensor	0.25	0.5	1 × 1 × 1	8	12	
Anemometer	0.5	1.2	1 × 1 × 2	8	2400	Samples 10 times/ second for twelve 20 second periods

Table 21

TV Lander Weight Summary

Internal equipment		86.2
Science	47.9 lbs.	
Electronics	38.3	
External equipment		18.8
Science	6.3	
Electronics	6.6	
Lights	1.0	
Flares and launch mechanism	4.4	
Heat sink material		15.0
Insulation (Min-K at 13.0#/ft.^{3})		25.0
Structure		92.1
Pressure vessel	32.0	
Internal mounting structure	17.6	
Soil sampler and TV mast mounting	2.4	
Insulation cover	11.0	
Body shock attenuator and mounting	1.3	
Fins and base cylinder	13.0	
Landing gear mounting brackets	1.2	
Release and installation hardware	13.6	
Parachute system		6.3
Chute, risers	4.0	
Harness, pyrotechnics, tractor rack	1.4	
Canister cover, ejection springs	.9	
Lander weight		248.4

Table 22

TV Lander Probe - Total Weight Summary

Lander weight		248.4 lbs.
Entry aeroshell (4 1/2' dia. 50° cone 1/2 angle)		98.0
Structure	56.0	
Heat shield forward	30.0	
Base heat shield	12.0	
Pyrotechnic and Delta structure for staging		18.0
Despin system		3.0
Entry weight:		367.4
Contingency (10%)		37.0
Adapter, separation mechanics, pyrotechnics		30.0
Contingency (10%)		3.0
Inertia balance weights (to achieve required roll inertia)		48.0
Contingency (10%)		4.8
		490.2
Spacecraft		266.0
Total system weight		756.2
Launch capability (1981 mission)		825.0 lbs.
Margin		68.8 lbs.

mirror is similar to that of the Surveyor TV camera and can be rotated around the boom axis and in elevation to scan the area around the lander from the horizon to about 1 ft. away from the surface point below the camera. The same gravity sensor that is used for pointing the high gain communication antenna to the zenith may be used to adjust the TV mirror system to compensate for lander tilt. By means of relay lenses and reflective optics the light from the image area is conducted to the inside of the lander and focused on the face plate of the camera tube.

The natural illumination of the Venus surface will be known from the Venus meteorology probe mission prior to the 1981 surface inspection mission. With sufficient natural light the visual range could be as far as 15 km although our present knowledge indicates that there would be a complete diffusion of natural light which would destroy all contrast except for that achieved by color or reflectivity variations. Natural illumination will be used for pictures of regions more than 20 ft. away from the lander and the second picture in the assumed sequence, see Figure 34, covers the horizon. This picture is evaluated on Earth and provides information about the best range and azimuth angles for the following pictures. Updating of the imaging controller on the lander is accomplished by command from Earth.

If natural illumination is insufficient for imaging, small (0.5 lb.) self-propelled rocket flares will be launched, possibly from the periphery of the lander antenna to insure accurate launching, toward a point about 60 ft. above the imaged area and 200 ft. away from the lander. Near this point the rocket flares provide a bright 3 second light source and illuminate equally the areas of the outer pictures 2, 7, 10, 11 and 12 which extend to a distance of 300 ft. These images will have good shadows because of the large angle between camera and light source.

For imaging of areas less than 20 ft. away from the lander, image contrast can be enhanced by light sources attached to the lander. Thus, regardless of the amount of natural illumination, reflector-backed, filament, quartz lamps will be attached to the outside of the lander for use in illuminating near-by areas. The 100 watt lamp corresponding to the programmed picture is turned on for approximately 3 seconds and during part of this time the camera tube is exposed.

For petrographic microscope pictures of the sample which has been collected and brought into the pressure vessel for chemical analysis, the image from the microscope is conducted to the camera by fiber optics. The camera is optically coupled to the microscope by flipping a mirror into the light path.

8.3.2 Surface sample analysis.

Surface sample analysis involves sample collecting, TV microscope viewing of the sample, and vaporization of the sample for mass spectrometer analysis.

The sample collector, Figure 33, is placed within the root of one of the lander fins and comprises a small diameter, helical fluted spindle with a tungsten carbide tipped drill. Spindle rotation at 8 rps is caused by a percussive drive which delivers many short duration blows, causing the drill bit to generate small surface particles which are driven upward by friction between the rotating particles and the tube wall. Near the top of the spindle, centrifugal force puts the surface sample particles into an inclined tube and the particles drop into the sample air lock. Then the upper valve of the air lock is closed and the lower valve is opened. The particles then drop onto a sample transporter which supplies the petrographic microscope and the composition analyzer.

8.4 MISSION STATUS

This mission was developed as a company sponsored effort by Martin Marietta (Reference 6). It is the result of a short period, but intensive application of present scientific and technical knowledge to the problem of Venus surface inspection.

PART III

DISCUSSION

1. MISSION-SERIES OPTIMIZATION; RELIABILITY; LOW COST

Part II has presented a view of six different classes of Venus missions and has indicated the scope of their science return. It is clear that, despite the findings of the Mariner and Venera missions and Earth based measurements, the ratio of future science return to future effort is large for Venus by comparison with present exploration possibilities for other planets. It is apparent also that:

a. The mission series will involve many kinds of experiments from a number of different planet disciplines, many different experimenters, and a constant return of new Venus information over a number of years. Experiment planning and execution can be expected to benefit from more thought and to proceed more reliably and at less total cost.

b. The mission-series can be planned to allow maximum use of new knowledge in planning the next steps in Venus exploration; a cost reduction feature.

c. The mission-series can be planned to allow for a sensible failure rate, i.e. the use of exploratory measurements which may produce negative results, in order to substantially reduce total mission-series cost.

d. The mission-series can be planned for maximum use of commonality in order to increase the science return to cost ratio by means of lower construction and testing costs and increased reliability.

Additional reliability is a feature of those missions where the bus vehicle impacts the planet. Should the command capability to the bus be lost following the third bus course-correction, an automatic probe-release program will carry out the mission substantially as planned. Should the automatic probe-release program fail, automatic release of the probes from the bus will take place as the bus enters the sensible Venus atmosphere and, although the intended probe placement will not be obtained, there will be a major science return.

2. INSTRUMENT STATUS

The principal task of this document was to show that it is possible to place, protect, operate, and communicate with the required instruments. Most of

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the means are straight-forward applications of present technology. Protection of optical windows from possible "frosting", for example, can be obtained by focusing an infra-red spot of aperture size on the window, by resistance heated apertures, or by maintaining a positive air pressure in front of the window from a filtered air source. With protection accomplished, use of most of the instruments—pressure gauges, temperature gauge, nephelometer, seismometer, radar, etc.—differs little, if any, from past use.

For some instruments, however, additional testing is required or mechanical and electrical adjuncts must be devised. Mass-spectrometer input leaks exist which will allow continuous operation at 1 atmosphere and in principle they can be used for several hours at 170 atmospheres and 750 K in a simple series leak-and-ballast-volume arrangement. The arrangement must be tested. If leak-inputs are unsuitable, then the principles of existing high-pressure metered-input gas systems must be adapted to Venus use; no intrinsic difficulty has been seen. For a few possibly more difficult instrument uses, such as a Venus surface chemical composition measurement, suitable techniques and high temperature, high pressure components have been identified.

In summary, most of the desired science return of the mission series can be obtained with existing instruments. The necessary principles, techniques, and components appear to be available for obtaining all of the desired measurements.

3. SYSTEM STATUS

No system design, construction, or operation difficulties have been identified which would render any significant part of the mission-series hopelessly inefficient or would place its achievement in jeopardy. The degree to which the balloon inflation system can be reduced in weight, for example, will not determine whether or not a circulation mission is possible but whether 6 or 9 balloons will be released at Venus.

While not all conceivable contingencies will be able to be coped with—a seismometer probe stuck on edge between two rocks will fail to communicate to Earth—the proposed mission-series is conceived so as to suffer least, on the whole, from the intrusion of unlikely or unanticipated events.

4. MISSION-SERIES RATIONALE

As we examine the science return contemplated by the mission-series it is clear that although the measurements of Mariners 2 and 5, and Veneras 4, 5, and 6, past Earth based measurements, and the anticipated Venus-Mercury

fly-by results, have begun to excite the interest of geophysicists, and represent a valuable starting point, most of our measurement and understanding of Venus will remain to be accomplished.

It is expected that additional work on the first and second phases of the study will both expand and strengthen the rationale for the Venus mission-series. The work to date indicates a low-cost, efficient, reliable, sustained, high yield program involving many disciplines and many scientists, suitable for international participation, making maximum use of past Venus measurements and future Earth based inspection of Venus.

The nearness of Venus to Earth and to the Sun, its similarity in size to Earth, its possible similarity in origin, its dense atmosphere, its uniform cloud cover, its high surface temperature, and its slow rotation make Venus the most interesting and at the same time the strangest of the inner planets. A growing number of scientists believe that the scientific importance of Venus is at least equal to that of any other object in the solar system and that to explore and understand this planet and its atmosphere should be an item of continuing priority in any planet exploration progress. They predict that the study of Venus will force a re-examination of ideas; a return from Earth-developed, difficulty-avoiding, semi-empirical approaches to an application of fundamental laws which will have a striking effect upon Geophysics in future years, and lead to a much better understanding of Earth's atmosphere and its present problems.

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APPENDIX A

GSFC No. 3609 Venus Model Atmosphere

95% CO_2, 5% N_2		g_0 = 8.88 m/s^2 at 6050 KM Radius		
Radius, KM	Z, KM	T, °K	P, ATM	Dens, G/M^3
6045	-5	776	1.47D 02	9.99D 04
6050	0	738	1.09D 02	7.74D 04
6055	5	700	7.88D 01	5.92D 04
6060	10	663	5.63D 01	4.46D 04
6063	13	641	4.55D 01	3.74D 04
6065	15	624	3.94D 01	3.32D 04
6070	20	584	2.70D 01	2.43D 04
6075	25	544	1.8D 01	1.74D 04
6080	30	499	1.16D 01	1.22D 04
6085	35	455	7.19	8.31D 03
6090	40	411	4.25	5.44D 03
6095	45	367	2.37	3.40D 03
6100	50	327	1.23	1.98D 03
6105	55	292	5.91D-01	1.07D 03
6110	60	262	2.61D-01	5.24D 02
6115	65	249	1.07D-01	2.26D 02
6120	70	241	4.29D-02	9.34D 01
6125	75	233	1.66D-02	3.74D 01
6130	80	225	6.25D-03	1.46D 01
6135	85	217	2.27D-03	5.49
6140	90	209	7.98D-04	2
6145	95	201	2.69D-04	7.02D-01
6150	100	193	8.71D-05	2.36D-01
6155	105	185	2.69D-05	7.62D-02
6160	110	177	7.92D-06	2.34D-02
6165	115	170	2.21D-06	6.84D-03
6170	120	190	6.44D-07	1.78D-03
6175	125	210	2.13D-07	5.33D-04
6180	130	210	7.42D-08	1.86D-04
6185	135	210	2.59D-08	6.50D-05
6190	140	244	9.87D-09	2.10D-05
6195	145	280	4.31D-09	7.94D-06
6200	150	336	2.15D-09	3.26D-06
6205	155	392	1.20D-09	1.55D-06
6210	160	471	7.36D-10	7.82D-07
6215	165	551	4.90D-10	4.42D-07

95% CO_2, 5% N_2		g_0 = 8.88 m/s^2 at 6050 KM Radius		
Radius, KM	Z, KM	T, °K	P, ATM	Dens, G/M^3
6220	170	630	3.46D-10	2.70D-07
6225	175	710	2.56D-10	1.76D-07
6230	180	710	1.93D-10	1.31D-07
6235	185	710	1.46D-10	9.81D-08
6240	190	710	1.11D-10	7.37D-08
6245	195	710	8.43D-11	5.55D-08
6250	200	710	6.44D-11	4.20D-08
6255	205	710	4.93D-11	3.18D-08
6260	210	710	3.79D-11	2.42D-08
6265	215	710	2.93D-11	1.84D-08
6270	220	710	2.26D-11	1.41D-08
6275	225	710	1.76D-11	1.08D-08
6280	230	710	1.37D-11	8.35D-09
6285	235	710	1.07D-11	6.45D-09
6290	240	710	8.40D-12	5.00D-09
6295	245	710	6.60D-12	3.89D-09
6300	250	710	5.21D-12	3.03D-09
6305	255	710	4.12D-12	2.37D-09
6310	260	710	3.27D-12	1.86D-09
6315	265	710	2.61D-12	1.47D-09
6320	270	710	2.08D-12	1.16D-09
6325	275	710	1.67D-12	9.18D-10
6330	280	710	1.34D-12	7.29D-10
6335	285	710	1.08D-12	5.81D-10
6340	290	710	8.69D-13	4.64D-10
6345	295	710	7.03D-13	3.72D-10
6350	300	710	5.70D-13	2.98D-10
6355	305	710	4.64D-13	2.41D-10
6360	310	710	3.78D-13	1.94D-10
6365	315	710	3.08D-13	1.57D-10
6370	320	710	2.52D-13	1.28D-10
6375	325	710	2.06D-13	1.04D-10
6380	330	710	1.69D-13	8.50D-11
6385	335	710	1.39D-13	6.95D-11
6390	340	710	1.14D-13	5.69D-11
6395	345	710	9.40D-14	4.66D-11
6400	350	710	7.74D-14	3.82D-11
6405	355	710	6.38D-14	3.15D-11
6410	360	710	5.26D-14	2.59D-11

95% CO_2, 5% N_2		g_0 = 8.88 m/s^2 at 6050 KM Radius		
Radius, KM	Z, KM	T, °K	P, ATM	Dens, G/M^3
6415	365	710	4.35D-14	2.13D-11
6420	370	710	3.59D-14	1.76D-11
6425	375	710	2.97D-14	1.45D-11
6430	380	710	2.46D-14	1.20D-11
6435	385	710	2.04D-14	9.90D-12
6440	390	710	1.69D-14	8.19D-12
6445	395	710	1.40D-14	6.78D-12
6450	400	710	1.16D-14	5.61D-12
6455	405	710	9.63D-15	4.66D-12
6460	410	710	8.00D-15	3.87D-12
6465	415	710	6.64D-15	3.21D-12
6470	420	710	5.52D-15	2.67D-12
6475	425	710	4.59D-15	2.22D-12
6480	430	710	3.81D-15	1.84D-12
6485	435	710	3.17D-15	1.53D-12
6490	440	710	2.64D-15	1.27D-12
6495	445	710	2.20D-15	1.06D-12
6500	450	710	1.83D-15	8.82D-13

APPENDIX B

Orbit Characteristics for Venus

Apoapsis altitude (km)	Periapsis altitude (km)	Orbit period (hrs.)	Ref. alt. (km)	% of Period below ref. alt.	Path angle below ref. alt. (degrees)
10,000	1000	3.81	2000	13.8	112.3
			5000	34.1	213.8
	400	3.7	1000	9.5	88.6
			2000	16.6	140.8
			5000	35.1	226.2
	200	3.61	1000	10.79	102.14
			2000	17.36	148.96
			5000	35.36	230.02
20,000	1000	6.52	2000	6.6	97.5
			5000	15.4	179.2
			10,000	30.1	245.0
	400	6.34	1000	4.6	78.6
			5000	15.9	192.2
	200	6.18	1000	5.2	90.92
			2000	8.28	131.38
			5000	16.05	196.44
			10,000	30.17	254.70
30,000	1000	9.70	2000	4.1	92.6
			5000	9.5	168.5
			10,000	17.8	226.4
	400	9.45	1000	2.8	75.1
			5000	9.7	181.6
	200	9.22	1000	3.23	86.98
			2000	5.12	125.36
			5000	9.80	185.84
			10,000	17.01	239.02
			20,000	39.49	294.06
40,000	1000	13.4	2000	2.85	90.08
			5000	6.57	163.30
			10,000	12.20	217.84
			20,000	25.81	272.52

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Apoapsis altitude (km)	Periapsis altitude (km)	Orbit period (hrs.)	Ref. alt. (km)	% of Period below ref. alt.	Path angle below ref. alt. (degrees)
	400	13.0	1000	1.99	73.34
			5000	6.75	176.36
			10,000	12.19	226.04
			20,000	25.67	277.16
	200	12.7	1000	2.26	84.98
			2000	3.58	122.30
			5000	6.80	180.60
			10,000	12.18	228.78
			20,000	25.62	278.72
50,000	1000	17.3	2000	2.14	88.58
			5000	4.91	160.20
			10,000	9.05	212.86
			20,000	18.71	263.92
	400	16.8	1000	1.49	72.26
			2000	2.57	113.16
			5000	5.05	173.22
			10,000	9.04	221.20
			20,000	18.58	268.94
	200	16.4	1000	1.69	83.76
			2000	2.68	120.44
			5000	5.08	177.46
			10,000	9.03	223.98
			20,000	18.53	270.62